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A Review of Australian and New Zealand Investigations on Aeronautical Fatigue During the Period April 1997 to March 1999

Colin Martin

DSTO-TN-0206

19990719 145

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Colin Martin

Airframes and Engines Division
Aeronautical and Maritime Research Laboratory

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ABSTRACT

This document has been prepared for presentation to the 26th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Bellevue, Washington USA on 12th and 13th July 1999. The review covers fatigue-related research programs as well as fatigue investigations on specific military and civil aircraft in research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 1997 to March 1999.

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Published by

DSTO Aeronautical and Maritime Research Laboratory
PO Box 4331
Melbourne Victoria 3001 Australia

Telephone: (03) 9626 7000
Fax: (03) 9626 7999
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AR-010-988
June 1999

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Executive Summary

The Australasian delegate to the International Committee on Aeronautical Fatigue (ICAF) is responsible for preparing a review of aeronautical fatigue work in Australia and New Zealand for presentation at the biennial ICAF conference. The Aeronautical and Maritime Research Laboratory (AMRL) has traditionally provided the Australasian delegate to ICAF and publishes the review as a DSTO document. This document later forms a chapter of the ICAF conference minutes published by the conference host nation. The format of the review reflects ICAF requirements.

Contents

8.1 INTRODUCTION

This review of Australian and New Zealand work in fields relating to aeronautical fatigue in the period 1997 to 1999 comprises inputs from the organisations listed below. The author acknowledges these contributions with appreciation. Enquiries should be addressed to the person identified against the item of interest. The postal addresses are listed below:

AMRL	Aeronautical and Maritime Research Laboratory, GPO Box 4331 Melbourne, Victoria 3001, Australia
DOTSE	Defence Operational Technology Support Establishment, Auckland, New Zealand Private Bag 32901
Monash University	Department of Mechanical Engineering Wellington Road, Clayton, Victoria, Australia
University of Sydney	Department of Mechanical Engineering Sydney University NSW 2006 Australia
Aerostructures Australia	Level 14, 222 Kingsway, South Melbourne Victoria 3205, Australia

8.2 FATIGUE PROGRAMS ON MILITARY AIRCRAFT

8.2.1 Australian F/A-18 Aft Fuselage and Empennage Fatigue Test (Paul White AMRL)

Testing of the F/A-18 aft fuselage and empennage has been conducted in two phases. Phase 1 represented RAAF and CF service loading experienced prior to the installation of aerodynamic fences on the wing leading edge extensions (LEX) of fleet aircraft to reduce buffet loading. Phase 2 loading represents post-LEX fence operations with reduced fin buffet.

A total of 1270 Phase 1 spectrum flight hours SFH were applied to the test article before Phase 2 loading was started. Currently the test is at 3661 SFH, having completed 2391 SFH of post-LEX fence loading.

The test loading system simulates the buffet loads experienced in flight using electrodynamic shakers, and manoeuvre loading using pneumatic actuators specifically developed for the test. The dynamic loading aims to reproduce the accelerations at the tips of the fins and horizontal stabilators measured during extensive flight testing . To check that the test loading is reproducing the correct strain distributions at the major structural locations a detailed fatigue comparison was carried out to compare the fatigue damage measured by strain gauges on the test article with strain gauge results taken from flight trials of the aircraft. These results showed that in general the strains were able to be reproduced on the test article. Where differences occurred the strains response indicated the applied test loads were slightly more severe than flight.

Inspections are focused on the aft fuselage using a variety of different methods. Eddy current for aluminium and titanium areas and magnetic particle for inspection of the steel stabilator spindle. The fin consists of aluminium ribs and spars with composite skins.
X-ray inspection is used to inspect the internal fin structure and in addition, a relatively new method of inspection of the aluminium spars under fasteners is being trialed using Low Frequency Eddy Current Array which has been developed by Northrop Grumman Corporation. Ultrasonic inspection is used to detect delaminations and core disbonds in the horizontal stabilators.

Fatigue cracking has been detected in the starboard Y580 and port Y590 fin stub attachments. A back to back steel bathtub fittings was installed on the Y580 stub to provide a full life fix. Blending and shot peening was used for the Y590 stub crack as an interim repair. It is anticipated that further cracking will occur in this area.

Testing will continue to 18000 SFH, with the possibility of testing further using dynamic loading only, to speed up testing. A full teardown of the structure will be undertaken at the end of testing.

8.2.2 F/A-18 FS488 Bulkhead Fatigue Test (David Graham Murray Stimson AMRL)

AMRL has been testing one of the wing-carry-through bulkheads for the F/A-18, the fuselage station 488 (FS488) bulkhead, in support of the Canadian IFOSTP centre fuselage test. The aim of the test was to determine if the OEM modified critical section of the bulkhead required additional rework, to achieve full life under RAAF and CF operating conditions. The critical location is beneath the lower wing pin lug, and the modification consisted of re-profiling a flange radius and also the outer surface of the bulkhead at a longeron recess to reduce the local Kt. The surface was then re-shot peened.

The test has achieved 34,500 spectrum flight hours (SFH). Many failures occurred outside the test area where loading was non-representative, but only one failure occurred at, 18,500 SFH, in the test section. This premature failure occurred as a result of the removal of the beneficial compressive layer developed by the shot peening process. The surface material was removed during the re-application of a strip strain gauge on the bulkhead flange. It was quite evident from the fractographic analysis that peening did indeed impede crack grow at the surfaces which still retained shot peening.

During testing of the bulkhead an Acoustic Emission System was used to monitor one side of the test article, while a new crack detection system, developed by Structural Monitoring Systems, was used on the other. This system uses a specially developed flexible pad containing an array of surface capillaries. The pad is bonded to the test area, and a small vacuum is applied to alternate capillaries. If a crack is present, the capillaries are breached, and the vacuum is lost.

8.2.3 F/A-18 Usage Monitoring (L. Molent)

Requirements for a Service Life Monitoring Program

Military aircraft will continue to be procured with fatigue usage monitoring systems fitted. These form one part of the Service Life Monitoring Program (SLMP) for the aircraft fleet. The requirement for usage monitoring is mandated by several design regulations eg Def Stan 00-970 and USAF Military Standard.

These standards specify the requirement and objective of implementing Individual Aircraft Tracking (IAT) and/or Operational Loads Monitoring (OLM) programs. Although the standards specify the need for a data recording system, albeit with differing emphasis, they allow much scope for variation in the implementation of the systems and interpretation of the data.

A limited literature review [1], has found many examples of the implementation of usage monitoring hardware. However there is a lack of reporting on the details of the philosophy used for selection of the hardware and the utilisation of the collected data. It is therefore not surprising that different philosophies are used to varying degrees of success by operators to monitor the accumulation of fatigue damage for individual aircraft types.

Work has been carried out at AMRL to developed a set of generic requirements for achieving a reliable monitoring system for primary load carrying (safety of flight) structural members of agile aircraft. This work built on the unified monitoring philosophy previously proposed by the author [2, 3, 4], and stems from a recently completed review of the RAAF's F/A-18 SLMP [5, 6].

It is hoped that with further development and peer review these requirements will eventually form part of a guidance document for SLMP implementation. This will assist the fleet manager in avoiding some of the problems associated with SLMPs and thus aid in achieving the full potential of in-service usage monitoring.

Further work has highlighted the challenge presented by the usage monitoring of vertical tails subjected to severe buffet environments [7]. This work concentrated on the requirements for highly agile aircraft, like the F/A-18.

Stress Estimation at Notches

To maintain the continued airworthiness of military aircraft it is essential that the fatigue behaviour of components subjected to complex multi-axial stress conditions be both understood and predicted as accurately as possible. Numerous fatigue failure criteria ranging from the purely empirical to the theoretical have been proposed. These rely on the accurate estimation of the stress and strain state at fatigue critical locations, normally associated with stress concentrations in the form of notches. Work was completed [8] that examined the relative advantages of the Neuber and the Glinka methods for calculating localised notch strains. These former techniques are at the core of several sequence accountable crack initiation prediction models, some of which are used in the life assessment of RAAF F/A-18 aircraft. Thus the accuracy of these techniques directly impact on the estimated fatigue life of these aircraft. The accuracy of the Neuber and Glinka methods was assessed by comparison with the results of detailed finite element analysis. This analysis confirmed that the appropriateness of these techniques was dependent upon the stress state (plane stress or plane strain) at the notch root.

8.2.4 F-111 Aircraft Structural Integrity Sole Operator Program (K. Watters – AMRL)

Following the retirement of the US fleet, Australia is now the sole operator of the F-111, and plans to operate the aircraft to 2020. A large program of work, known as the F-111 Aircraft Structural Integrity Sole Operator Program (ASI SOP), has been developed to meet the challenges of maintaining safety and availability of our fleet [9].

Five significant tasks have been placed on the F-111 OEM, Lockheed Martin Tactical Aircraft Systems (LMTAS) at Fort Worth in the US. Attached Australian engineers and LMTAS engineers jointly staff those tasks. The outputs of those tasks will be: a finite element internal loads model of the F-111 fuselage and wings; a comprehensive report on the durability and damage tolerance assessment (DADTA) of the Australian F-111; an extension of DADTA capability to model multi-site crack initiation; a report on the basis and status of external loads characterisation of the F-111; and a report documenting the set of Structurally Significant Items (SSI). A significant outcome of the LMTAS tasks will be the transfer of F-111 structural engineering capability from the OEM to Australia. The most important transfer will be the capability to conduct future DADTAs of the F-111 in Australia. The LMTAS tasks commenced in July 1998 and are scheduled to finish in August 2000.

An ex-USAF F-111 fuselage with high service hours will be subjected to a teardown inspection in Australia. Planning for the teardown commenced in July 1998 and the infrastructure is now in place. The teardown fuselage is expected to be delivered from the US in August 1999 and the teardown will then commence.

An ex-RAAF F-111 wing with in excess of 5,000 hours and 25 years of service will be further 'aged' by fatigue loading in a test rig and will then be subjected to a teardown inspection. The wing test rig has been designed and is currently being manufactured. The wing test is expected to start in July 1999 and will take approximately 2 years. The ensuing teardown will then take a further 18 months.

The primary purpose of the fuselage and wing teardowns is to reveal any new degradation sites in the F-111 structure which might be manifest in the service fleet over the next 20 years. The teardowns will also

give an indication of the current state of degradation of the fleet, including any widespread degradation, which will provide a basis for projecting to the future state and making structural integrity management decisions.

Other work in progress on management of known fatigue and environmental degradation problems in the F-111 structure include: Using a FE model of the wing pivot fitting to investigate sites prone to fatigue cracking, caused primarily by residual stresses left after cold proof load testing. Investigating the disbond and corrosion degradation in bonded honeycomb panels. The investigation will focus on the structural significance of that degradation and the repair and replacement options.

A broad program of corrosion research is being conducted with relevance to F-111. This work examines the corrosion susceptibility and the corrosion rates throughout the structure, and a way of incorporating the effect of corrosion in DADTA as a precipitator of fatigue. A research program on NDI Probability of Detection, particularly using magnetic rubber inspection of D6ac steel is also in progress.

8.2.5 Damage Tolerance Analysis of F-111 Overwing Longeron Repair (R. D. Locket & D. A. Rees – Aerostructures)

Following the discovery and repair of unrecorded rework to a cover panel fastener hole on an F-111C overwing longeron, concerns were raised over the damage tolerance of the repair and a relatively short interim inspection interval was imposed. A detailed stress and damage tolerance analysis was carried out to determine appropriate remedial action and calculate an inspection interval based on RAAF usage.

A detailed three dimensional stress analysis of the longeron defect was carried out using the finite element method and critical locations identified. Measured RAAF normal acceleration spectra and data from previous F-111 DADTA studies were used to develop stress spectra for the defect location. Crack growth models for the critical locations were developed using linear elastic fracture mechanics and a Wheeler retardation model. Sensitivity studies were conducted to investigate the effects of spectrum variability and crack growth retardation.

The study led to a recommendation that the repeat inspection interval be increased by a factor of ten, thus enabling alignment of the inspection with a major servicing. No further repair action was recommended.

8.2.6 Fatigue and Damage Tolerance Analysis of F-111 Nose Landing Gear Axle Adapter (R. P. Armitage & D. A. Rees – Aerostructures)

In-service fatigue cracking in the nose landing gear axle adapters was discovered on a number of F-111 aircraft. An investigation was carried out to identify the cause of the cracking and determine appropriate preventative and repair action.

A detailed three dimensional stress analysis was carried out using the finite element method. Critical design ground conditions for the axle adapter were analysed. The study indicated that the cracking could be attributed to specific ground load conditions. The fatigue life was estimated using a cumulative damage model based on the stress analysis results and a load spectrum modified for RAAF operations. The calculated life was found to be consistent with in-service cracking.

Further analysis indicated that significant stress reductions could be achieved by blending of material from the critical location. Several progressive rework profiles were proposed and the fatigue lives estimated.

A damage tolerance analysis was carried out to establish appropriate inspection and/or rework intervals. A conservative crack growth model based on linear elastic fracture mechanics was developed. A retardation model was not used. A management strategy was developed to ensure safe and economical operations through to the planned withdrawal date.

8.2.7 Full Scale Fatigue Test of the Pilatus PC9/A Trainer Aircraft (Raymond Parker and Ian Anderson AMRL)

The Aeronautical and Maritime Research Laboratory is conducting a full-scale fatigue test of a Pilatus PC9/A trainer aircraft. The aim of the test is to define the safe fatigue life of the major structural elements when subjected to usage typical of Royal Australian Air Force (RAAF) operations.

Test loads representative of those experienced in flight are applied to the wings, tailplane, fin, fuselage, engine mount frame and main landing gear support structure. Loads are applied via a system of closed-loop servo hydraulic jacks and fixed reactions via a digital controller/data acquisition system. The overall test load sequence comprises manoeuvre sequences, spin sequences and ground load sequences. Manoeuvre loads are applied to the wings, tailplane, fin fuselage and engine mount frame. Spin loads are applied to the tailplane and fin. Ground loads are applied to the main landing gear, and are applied to test the main landing gear support structure contained within the wing.

The main input into the derivation of manoeuvre load sequences was strain sequences recorded at various structural locations, while the aircraft was flying sorties typical of RAAF operations. A total of 28 representative sorties were flown.
Spin loads, which were found to be characterised by vibration or buffet loading, were also based on flight recorded strain sequence data. Additional flight-testing was performed with empennage strains recorded at a higher sampling rate than used for manoeuvre flights.

Ground loads were derived using a standardised approach based on UK fatigue design requirements for landing gear.

To relate strain data recorded during manoeuvre flying and spins to loads data, an extensive ground load calibration of the flight test aircraft was performed. Load equations relating load at nominated load reference stations to strain were developed, and flight recorded strain sequence data were converted to load sequence data.
The resulting load data for manoeuvre sequence and spin sequences were processed to remove non fatigue damaging events, and to arrive at a sequence of actuator test loads.
Manoeuvre sequences, spin sequences and ground load sequences were combined into a master sequence. The overall Nz exceedence counts, the number of spins and the number of take-off and landings matched that of average fleet operations at the RAAF Central Flying school.

Extensive test loads checking was performed prior to commencing formal testing, which resulted in some improvements to the test loading arrangement and to the test load sequence. Formal test cycling began in January 1996. The target of 50,000 Simulated Flight Hours (SFH) was achieved by February 1999. Structural damage was discovered at at a number of locations. Repairs and modifications have been developed and fitted to the test article in consultation with the manufacturer to ensure adequate fatigue life.

The test has provided considerable usefull information to aid the manufacturer and the RAAF to achieve the target fatigue life.

8.2.8 P3 Flight Loads Test Program
(M. Houston - Aerostructures, P. Jackson – AMRL, A. Last – RAAF)

The United States and other countries are planning to extend the service life of the Lockheed P-3 Maritime Patrol aircraft by at least 50%. In order to substantiate this extension, the United States Navy (USN) has embarked on an ambitious Service Life Assessment Programme (SLAP), a component of which is a Full Scale Fatigue Test (FSFT). FSFT loads will be derived using analytical and wind tunnel methods, as a comprehensive loads development flight test program is not viable and little data is available from early demonstration testing. However, the USN recognises the need to verify the calculated loads and has included a loads verification task in the FSFT Statement of Work (SOW).

In response to an invitation to participate in SLAP, the RAAF offered to prepare and conduct a P-3C Flight Loads Test Programme (FLTP) that would be required to support the FSFT loads verification task. The value of such an undertaking to the RAAF was maximised by programming the collection of data necessary to validate and refine the automated usage monitoring system and Service Life Monitoring Programme (SLMP). The P-3 FLTP took place between November 1997 and March 1999 and was conducted from the RAAF Base Edinburgh, South Australia.

The flight test aircraft was instrumented with approximately 100 strain based and 40 flight parameter measurands. A ground calibration of the loads instrumentation was performed prior to the commencement of the flight program. To simplify the calibration task in recognition of strict schedule constraints, a low load calibration technique validated by Lockheed during P-3 development was implemented. The Flight Test Matrix required the conduct of a set of symmetric and asymmetric manoeuvres at a variety of Points-

In-The-Sky (PITS), weight, and centre-of-gravity combinations. Loads equations derived following the ground calibration were applied to produce flight loads data for use in SLAP and SLMP development.

8.2.9 DHC-4 Caribou Wing Teardown (P. Jackson – AMRL)

The RAAF DHC-4 Caribou fleet is approaching its structural safe life originally defined by a full scale fatigue test conducted by De Havillands Canada in the early 1960's. A replacement for the Caribou fleet is being considered by the RAAF, however it is expected that a number of aircraft will exceed their safe life limit prior to replacement. Airworthiness assurance will transition from safe life to safety by inspection methodology with an inspection program based on the fatigue test results and in-service experience. To support the transition to inspection based airworthiness AMRL conducted a teardown of a Caribou centre and outer wing that had reached its safe life limit. The purpose of the teardown was to provide assurance that the results of the original fatigue test over 30 years ago were still applicable, and to expose any significant structural deterioration that had not been dealt with during routine aircraft maintenance. Corrosion dominated the teardown findings that nevertheless showed that the caribou wing structure currently remains free of significant defects.

8.2.10 Operation of DHC4 Caribou Beyond the Safe Life Limit (D. A. Rees – Aerostructures)

The Caribou has been in service with the RAAF for 35 years. The aircraft is currently managed on a safe life basis using the Caribou Service Life Monitoring Program (CARSLMP) and many aircraft are soon to reach their safe life limit. It was originally intended by the manufacturer that operations beyond the safe life limit be managed by a series of structural inspections based on demonstrated fail safe characteristics, as determined by the certification requirements of the 1950s. To assist structural management planning an investigation to determine an appropriate certification basis for continued operations has begun.

A survey of fleet structural condition data was carried out to establish known fatigue critical locations or problems. RAAF defects and non-standard repair records were analysed. Data was compared to fatigue test and wing teardown results. An assessment of the CARSLMP was also undertaken to review the validity and inherent conservatism in the safe life calculation. RAAF usage data, which is the major input into the CARSLMP, was also reviewed.

The requirements for a safety by inspection program based on modern certification standards are also being established and feasibility with regard to the Caribou airframe assessed. A revised aircraft structural integrity management plan for continued operations is being developed.

8.2.11 RAAF C-130E Service Life Analysis (C. K. Rider – Aerostructures)

The delayed delivery of the C-130J model to the RAAF has led to a possible extension of the planned withdrawal date of the C-130E aircraft. A re-assessment of the service life of the C-130E aircraft was carried out. As RAAF aircraft are fitted with original outer wings, the assessment concentrated on these components.

Current RAAF usage of the C-130E aircraft was examined by analysis of mission mix and fatigue meter data. Mission mix data was compared with RAAF C-130E usage which was assumed to be representative for outer wing service life assessments carried out by Lockheed Georgia in 1978 and 1984. The relative severity of training and transport missions from an assessment of fatigue meter data was compared with previous mission severity factors. The major influence of cargo weight on earlier fatigue damage calculations for the lower wing surface was reviewed, and applied to a similar calculation for current usage. Outer wing structural condition data was also reviewed. Stress corrosion defect data for the outer wing engine dry bay areas were systematically organised to document the variability between wings and

the upper/lower surface differences. An attempt to compare RAAF usage with C-130 full scale fatigue test loading and results was also made. It was concluded that the tests were severe in comparison to RAAF usage, but a reliable estimate of equivalent RAAF hours was not possible due to the block loading nature of the test spectra. Conclusions and recommendations with regard to outer wing service life were made.

8.2.12 Damage Tolerance Analysis of C-130E Outer Wing Repairs (D. A. Rees – Aerostructures)

Multiple repairs due to riser pad cracking in the lower skin panels of the C-130E outer wings led to concern over structural integrity. An investigation into the static strength and damage tolerance of a wing skin panel was undertaken to evaluate the implications of multiple repairs. Static strength margins of safety were determined and critical locations identified. A two dimensional finite element model was developed to quantify the effects of modifications and repairs on the skin panel stresses. In particular, stress concentration effects at the ends of riser channel doubler repairs were investigated. Residual strength evaluation considered two likely cracking scenarios. The most critical was a through crack growing beneath a fractured riser. The critical crack length was determined for this condition. Crack growth was predicted for several cases based on recent measured RAAF usage data and typical material crack growth data. Simultaneous cracking from multiple holes at the end of adjacent channel repairs (multi-site damage) was also considered and this condition was found to be the most critical. The sensitivity of crack growth rates to material data, usage severity and fracture toughness was addressed.

Inspection intervals were determined and repeat inspections recommended for aircraft fitted with these repairs. Inspection intervals determined in this study were consistent with intervals previously specified for similar configurations by the aircraft manufacturer. The significance of multi-site damage scenarios in aging aircraft structure was highlighted by this investigation.

8.2.13 B707 Structural Life of Type Studies (P. Jackson – AMRL)

The RAAF fleet of B707 tanker/transport aircraft have all reached the original design life of the aircraft (20,000 flights or 20 years service) and are now subject to the Boeing developed aging aircraft related program of structural inspections and mandatory modifications. The RAAF has recently commissioned AMRL to conduct a study on the structural airworthiness requirements of continued operation of the B707 fleet. The RAAF aircraft are of a similar age and have a similar commercial operator background to the aircraft being purchased and converted by the USAF for JSTARS operations. The JSTARS conversion program includes a very significant refurbishment effort to combat structural corrosion and to complete the aging aircraft program requirements. In addition, the USAF conducted a lower wing teardown program on two high life aircraft and found evidence of widespread fatigue damage in the lower wing skin/stringer panels. An accompanying USAF structural risk assessment showed that the panels would need to be replaced to provide both airworthiness assurance and to limit future aircraft maintenance down time. The RAAF, QANTAS and AMRL have now commenced a number on fact finding activities on the RAAF B707 aircraft including structural condition assessments based on JSTARS experience, sampling inspections of the wing lower skin panels and the conduct of a structural risk assessment, plus a review of in-service usage history. Whilst not yet complete the work has revealed similar corrosion and fatigue damage to that found by the USAF. The studies are expected to be completed by the middle of 1999 at which point the scope of any structural refurbishment will be determined.

8.2.14 Revision of RAAF B707 Damage Tolerance Analysis (V. Kowalenko – Aerostructures)

RAAF B707 aircraft were modified between 1988 and 1992 so that they could perform air-to air refuelling missions. The modification was certified to FAR 25 damage tolerance requirements and inspection intervals were promulgated based on an analysis by Israel Aircraft Industries. A revision of that damage tolerance analysis was carried out to take into account current RAAF usage.

The analysis used the standardised TWIST spectrum. The randomised sequences of flight load cycles were generated and groung-air-ground, landing impact, touch and go and taxi cycles inserted in accordance with recent RAAF usage data. The fracture analysis included operations with and without the refuelling pods fitted. The revised results showed increased upper surface crack growth intervals with little change for the lower surface. Revised inspection intervals were recommended.

8.2.15 Damage Tolerance Analysis of the Macchi MB326H Centre Section Lugs (D. A. Rees & A. Loh – Aerostructures)

Damage in the form of cracks and pitting corrosion has recently been discovered in a number of RAAF Macchi aircraft wing attachment lugs. An investigation into the damage tolerance of feasible repair options was undertaken to quantify the effects of lug rework on the structural inspection intervals currently used to manage the aircraft on a safety by inspection basis.

The current inspection program was determined on the basis of full scale testing of wing centre section caps undertaken at the Aeronautical and Maritime Research Laboratory (AMRL) in the mid 1990s. The tests employed a modified FALSTAFF (Fighter Aircraft Loading Standard For Fatigue) load spectrum and the test crack growth curves were used to develop inspection intervals. These inspection intervals were then promulgated in terms of the Macchi CTDS (Component Tracked Damage System) fatigue index.

An analytical study was undertaken to determine the effect of proposed lug repairs on the inspection interval requirements. A finite element analysis was carried out to determine stress gradients in critical locations of the wing attachment lugs. Crack growth under the modified FALSTAFF test spectrum was then calculated using Linear Elastic Fracture Mechanics (LEFM) and a Wheeler retardation model. The calculated crack growth curves were found to be in good agreement with experimental results. The analysis was then repeated for various rework configurations.

Recalculated inspection intervals based on the modified FALSTAFF spectrum were then compared with those for a load spectrum derived from measured RAAF normal acceleration exceedance spectra and found to be conservative. Although the inspection interval requirement is reduced for the repaired lugs, current inspection intervals are dictated by a more critical location on the lower boom of the centre section. This study concluded that no additional inspection requirements are necessary for the repaired lugs.

8.2.16 Helicopter Structural Integrity and HUMS in the Airframes and Engines Division (AED) of AMRL (by D. C. Lombardo, AMRL, unless otherwise indicated)

Helicopter structural integrity work within AED for the Australian Defence Force (ADF) continues to increase. A summary of the more significant tasks undertaken is given below.

Australian Regular Army, Sikorsky S-70A-9 Black Hawk

- Component Retirement Time Reassessment Project

At the time of the 1995 review, Sikorsky had just completed Phase 1 of the project; namely, the definition of a new usage spectrum and an impact study which showed that the new spectrum would result in greatly reduced component retirement times (CRTs). The data on which the new spectrum was based came from a pilot-questionnaire survey. Since then, Sikorsky has performed Phase 2 of the project, which entails a reassessment of the CRTs for 12 components. Apart from using the new spectrum in calculating the revised CRTs, Sikorsky also updated material SN data and re-examined the UH-60 Black Hawk flight loads database to remove irrelevant loads (e.g. loads due to firing Hellfire anti-tank missiles as the S-70A-9 does not have an attack role). The results of Phase 2 showed that components would have their CRTs reduced, but not to the extent suggested by the Phase 1 impact study. The new CRTs have been implemented.

- Fast Roping And Rappelling Device (FRRD)

AED provided advice on the fatigue design of the FRRD and performed the static and fatigue structural testing. The FRRD passed all tests and has now been installed in Australian Army Black Hawks.

- Internal Panel Cracking Flight Trial

The 1995 review mentioned that a Black Hawk had been instrumented so that an in-flight measurement program could be undertaken to determine the cause of internal panel cracking. The program was completed in early 1995, and the results showed that the cracking was due to the structure being too highly strained across the entire flight envelope. There were no particular aircraft configurations or flight conditions that made the problem worse. The only solution to solving the cracking was to redesign the internal panel. Sikorsky produced a redesigned panel, which has been installed in Australian Black Hawks.

- Quantitative Usage Assessment and Definition

The RAAF and the Army are to implement a quantitative program to measure Black Hawk usage, in-flight. The need for such a project stems from the desire to obtain more reliable data than the pilot-questionnaire data mentioned above. AED is assessing the possible options for such a program. These options range from using the flight data recorder, currently being installed in the Black Hawk fleet, as a usage monitor, to a comprehensive HUMS.

Eurocopter AS-350B Squirrel

- Tailboom Buckling Investigation

Hard landings under autorotation flight training have caused some tailboom buckling in Australian Defence Force Helicopter School (ADFHS) AS-350B Squirrel helicopters. However, when minor buckling occurred during a less severe landing, AED was asked to investigate the stresses in the critical section of the tailboom. A finite element (FE) analysis of the tailboom was undertaken and it showed that stresses in the tailboom might be sufficient to cause buckling during heavy landings. Subsequently, one ADFHS Squirrel was instrumented at AED and an in-flight loads measurement program undertaken to validate the FE analysis. The measured loads were found to confirm the FE analysis. The critical factor in determining the probability of tailboom buckling was found to be the rate of application of loading on the tailboom.

Helicopter Procurement Programs

- New Medium Helicopter for ANZAC Frigates

  The RAN announced (January 1997) the preferred tenderer for the supply of a helicopter for the ANZAC frigates. Kaman, with its SH-2G(A) Seasprite, won the tender over Westland with its Super Lynx. AED and Maritime Platforms Division provided advice on the relative merits of the two contenders in terms of structural, corrosion, engine, and drivetrain issues.

- Armed Reconnaissance Helicopter for Army

  The Army has defined its requirements for a new Armed Reconnaissance Helicopter (Project AIR 87) which is to provide reconnaissance and fire support capabilities. The Army asked advice from AED to help define the requirements. For example, should a separate helicopter type be used for the two roles, or can one helicopter type perform both and, if so, what compromises are involved? AED produced several reports providing advice in areas such as HUMS, the fitting of armament to civil airframes, assessment of usage, corrosion and marine deployment.

8.2.17 Health and Usage Monitoring Systems (HUMS) (G.F. Forsyth, AED)

Over the last few years DSTO provided technical input into an Australian Defence Organisation Working Party on HUMS. This working party produced a number of Guidance Papers and software to help evaluate the cost-benefit aspects of HUMS fitment. Several reports were produced looking at the benefits of HUMS for several Australian military helicopter types. A summary report, containing the recommendations of the Working Party, was also issued.

In February 1999, a military HUMS conference was held at AMRL. The conference, coordinated by AED, was attended by over 100 participants including over 40 foreign participants from Canada, France, Italy, Japan, New Zealand, South Africa, Switzerland, UK and USA. ADF participants were from all three services. Originally proposed as a workshop, the event grew to include two full days of formal papers as well as a demonstration of a Red Hawk helicopter. The published Proceedings (DSTO document DSTO-GD-0197) represents a good assessment of the current status of HUMS for military helicopters.

One area where DSTO is likely to have increasing interest involves collaboration on the evaluation of the growing number of trials of HUMS on military helicopters. One such trial is the DARPA Joint Advanced HUMS (JAHUMS) operational evaluation where Seahawk and Chinook HUMS are to be trialed in operational use.

8.3 FATIGUE OF CIVIL AIRCRAFT

8.3.1 Fatigue Implications to the IAI ASTRA 1125SP Aircraft in the Role of Navigation Aid Calibration in Australia (K.Jackson & B. Madley – Aerostructures)

Airservices Australia purchased an Israel Aircraft Industries 1125 Astra business jet in 1996. The aircraft was subsequently modified to enable it to undertake navigation aid calibration work in Australia. The type of flying performed in this role includes low and medium level flight with a considerable amount of manoeuvring and is significantly different from that assumed for a typical Astra mission profile. The Civil Aviation Safety Authority identified the need to determine the affect of this new role on the inspection requirements of the Astra aircraft.

Flight profiles were developed for the various missions carried out by the Astra in its new role. These flight profiles were then used to generate an Australian navigation aid calibration load spectrum for the aircraft. An Astra design load spectrum was also developed based on typical missions flown by business jets. The calculated spectra were compared with measured spectra obtained from the available literature both for business jets operating in a conventional role and for aircraft performing navigation aid calibration work in Australia in the 1970s. The navigation aid calibration load spectrum was found to be considerably more severe than the assumed design load spectrum.

A damage tolerance analysis was performed at selected locations on the aircraft wing and fuselage. The results demonstrated that inspection intervals need to be reduced by a factor of up to 2.1 at locations influenced by normal acceleration gust loading. A usage monitoring program which includes the measurement of normal acceleration was subsequently developed and implemented. This program will permit the Astra flight profiles and load spectra to be more accurately defined and will lead to a more accurate estimation of the total life and inspection intervals of the aircraft in a navigation aid calibration role in Australia.

8.4 FATIGUE RELATED RESEARCH PROGRAMS

8.4.1 Initial Defects and Short Fatigue Crack Growth in 7050 T7451 Thick Section Plate (G Clark and S Barter - AMRL)

Investigations into the size, type and nature of initiating flaws in 7050 aluminium alloy thick-section plate have commenced. The factors controlling the initiation and early growth of fatigue cracks in this material when subjected to representative aircraft loads and surface conditions is of great interest to the RAAF for the refinement of crack growth prediction models. These models are used to life the Australian F/A-18 aircraft, which contain numerous critical bulkheads manufactured from this material.

The testing is investigating:
(i) the nature and population of initial flaws in this material,
(ii) which of these flaws are most prone to developing into fatigue cracks, and
(iii) the way in which representative fatigue cracks in this material grow.

A particular focus is the interaction, and linking up of fatigue cracks during short crack growth, as this region of cracking has been shown to consume a very large portion of the crack life. As such, changes to this early crack growth can have a disproportional large effect on life. The linking process affects crack growth rates, but is not currently addressed in crack growth models. Testing is being carried out with loading representative of service, to produce the most useful data. In support a series of constant amplitude R-ratio tests on this material, with varying surface finishes, are also being conducted to produce base-line growth data.

The investigation involves acquiring data on flaws and cracks that have grown for only part of their total life, in the close proximity to other active cracks, allowing examination of the detailed interactions between the initial flaws and between developing cracks. In a parallel exercise, the details of the intrinsic defect type and size distribution in the material are being determined along with their locations.

The information derived will allow improved modeling of the early crack growth processes, and will support the modification of existing fatigue crack growth models. It will also provide a valuable base for improved statistical modeling of crack growth, for incorporation into reliability analyses.

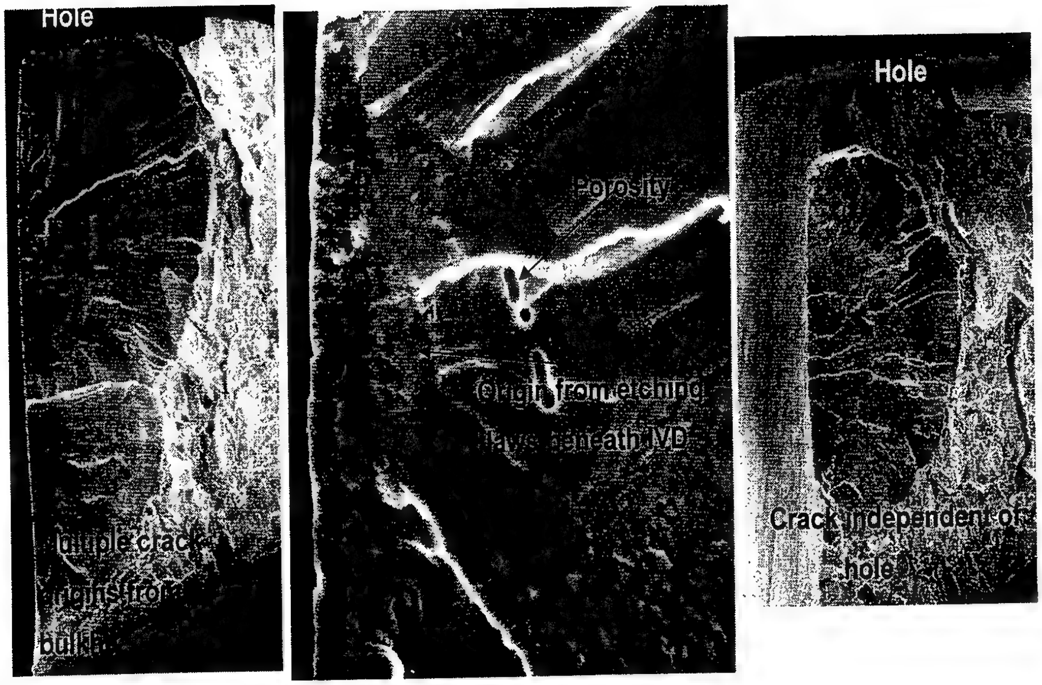


Figure 1: Initial defects in 705 aluminium. Showing porosity, and initiations at etch pits, and initiation remote from a stress concentrator.

8.4.2 Effects of Exfoliation Corrosion on Structural Integrity (G Clark and P Khan Sharp - AMRL)

The effects of exfoliation corrosion on fatigue cracking are being assessed using an Equivalent Initial Flaw Size approach. Initial results are based on laboratory-generated exfoliation, and confirm the large effect of corrosion flaw geometry on fatigue life. In both 7075 and 2024 alloys, specimens containing corrosion typically some 100-500 micrometres in depth led to a reduction in fatigue life of over two orders of magnitude. [Figure 2] The scale of this reduction indicates that for practical purposes, the dominant effect of such corrosion lies in its geometrical influence, and the details of any subsequent fatigue crack propagation are likely to exert only a second-order influence on overall life. In other words, any effect of the environment on fatigue crack propagation once corrosion of this kind has formed is likely to have negligible impact on structural integrity compared to the life-reducing effect of the initial corrosion defect.

Further analysis of the exfoliation defect geometry is focussing on testing simple geometrical representations of the corrosion geometry as crack starters, using closure-based fatigue crack growth rate models.

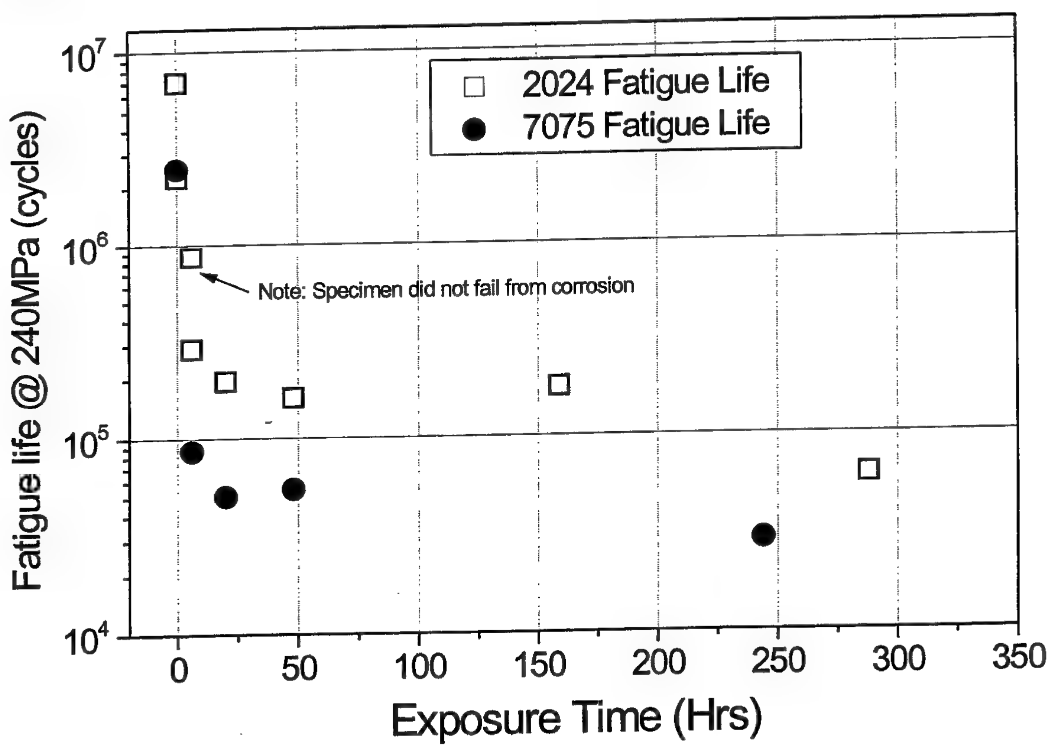


Figure 2

8.4.3 Effects of Laminar Corrosion Defects on Structural Integrity (G Clark and N Athiniotis - AMRL)

The structural significance of laminar corrosion defects (ie. stress-corrosion defects, usually growing on microstructure-controlled planes) is particularly difficult to assess. [see reference 10] There are several reasons for this;

(a) they are usually aligned with the principal loading direction in an aircraft component, and therefore appear to present a structural threat only by promoting instability/buckling. Conventional design does not address the possibility of such defects occurring and promoting such failure modes.
(b) the possibility clearly exists of a laminar defect acting as an initiator for fatigue cracks growing normal to the principal stresses, but such a configuration does not lend itself to conventional fatigue analysis.
(c) Laminar defects occur unpredictably, at holes or free surfaces; they are often influenced considerably by microstructure, material forming methods, and residual or fit-up stresses, rather than the stress concentrations which usually dominate fatigue. Identifying likely sites for laminar defects in an airframe is therefore considerably more difficult than identifying fatigue cracking "hot-spots". As a result it is extremely difficult to devise NDI programs to rule out the presence of laminar corrosion defects.

Investigation of the significance of laminar defects is progressing along two lines:

- Full-scale fatigue testing of assemblies likely to contain laminar corrosion defects. A number of tailplanes from aircraft which have seen extensive service, and which are thought likely to contain laminar corrosion defects are being tested to failure to determine: (a) whether any corrosion plays a part in the failure mode, (b) whether there is any evidence of fatigue associated with any laminar defects present, and (c) if so, the growth rate of the fatigue cracks, to allow determination of the EIFS for the laminar defects.
- A durability-based modelling approach, in which it is assumed that a fatigue crack initiates from a laminar defect when that defect encounters an intrinsic initial crack-like flaw oriented in a direction favourable to fatigue (ie. normal to the principal loading, and normal to the laminar defect). Analytical modelling of the stress intensity factors associated with fatigue cracking from this initial laminar flaw will allow a comparison with those expected in the absence of the laminar defect (ie. will identify the extent to which the laminar defect promotes fatigue). Initial results suggest that having an intrinsic initial flaw at the tip of the laminar corrosion defect is more damaging than the initial flaw itself (K for the combination being about 1.4x that for the isolated initial defect, but that it is less damaging than having that same flaw at a free surface. The effect of the laminar defect is almost independent of the laminar defect size, over the range normally encountered in service.

8.4.4 Effects of Pitting Corrosion in High Strength Steel on Structural Integrity (G Clark, C Loader and G Holden - AMRL)

To support F-111 operation for another two decades RAAF and AMRL are undertaking an extensive program to assess and ensure the structural integrity of the aircraft. With time, breakdown of the Cadmium plating on the high-strength steel carry-through structure is allowing the development of corrosion pits; such pits are known to be able to act as initiators of fatigue cracks, and must therefore be factored into the structural integrity analysis. Research is concentrating on two aspects:

- Survey and assessment of pitting in RAAF aircraft, by examination of parts from service aircraft, and identification of any associated fatigue cracking.

- Production in the laboratory of pitting which simulates the service case, followed by assessment of the growth rates of fatigue cracks from the simulated pitting. This will establish an EIFS for the various forms of pitting, and may identify critical characteristics of the pits which control fatigue crack development.

Early results include development of a method of producing pits in laboratory samples; these pits successfully replicate pits seen in RAAF aircraft, and are being used to produce fatigue simulation specimens. Detailed analysis of the pit geometries available to date from service aircraft indicate that pit

depths between 10 and 20 micrometres are common. The use of extreme-value statistics to extrapolate the pit samples to the RAAF fleet as a whole implies the presence of pit depths in the RAAF fleet up to 60 micrometres.

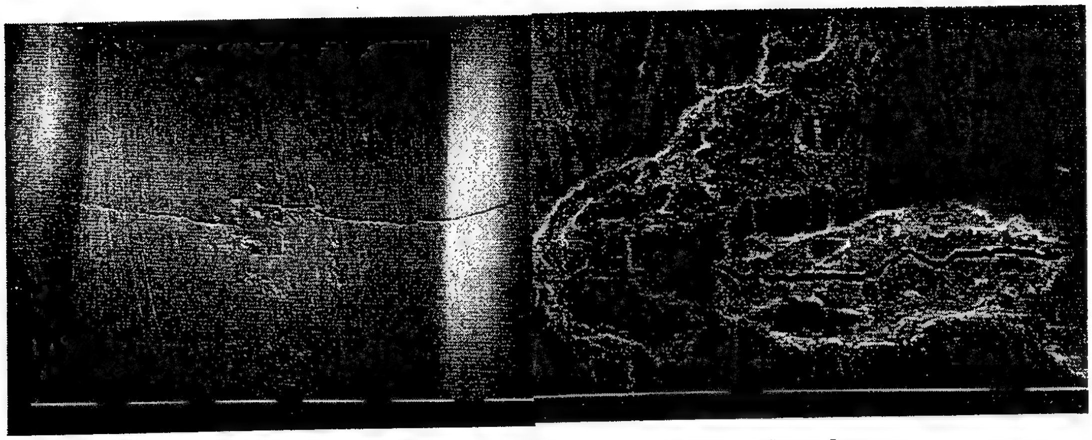


Figure 3: Fatigue cracks growing from corrosion pitting in high-strength steel.

8.4.5 Effects of Corrosion and Corrosion Preventives on Fatigue of Joints. (G Clark, P Khan Sharp, S Russo, B.R.W. Hinton – AMRL and Krishnakumar Shankar - ADFA)

The effect of corrosion on structural integrity is likely to be focussed on the fatigue performance of joints. The principal difficulty with assessing the effect of corrosion on joint fatigue is that the major effect is likely to be a change in failure locus (ie. failure site and/or failure mode), thus preventing simple comparisons of fatigue life. AMRL, in conjunction with the Australian Defence Force Academy, is studying the effect of corrosion and corrosion preventives on the failure locus in dogbone specimens. Early results indicate substantial changes in the initiation sites for fatigue, but with little change in the overall load transfer in the specimens.

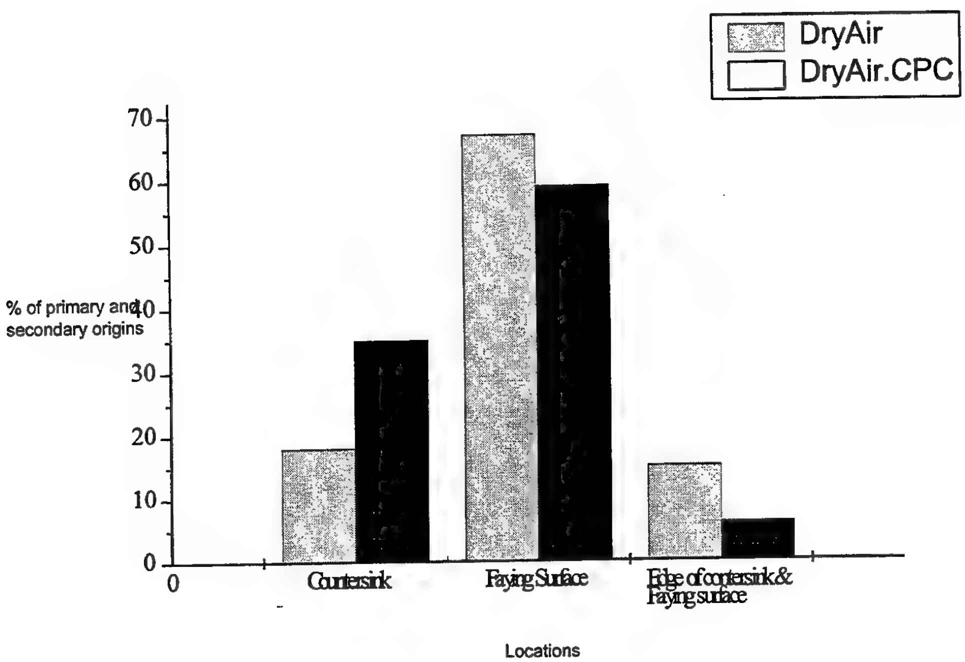


Figure 4: The shift in fatigue crack initiation site from the faying surfaces to within the countersink hole on the addition of CPC.

8.4.6 NDE of Corrosion (G Clark, S R Lamb, N Rajic - AMRL)

During an attachment of a DSTO scientist to NASA, the focus of research was development of a high-speed thermographic inspection system, with particular emphasis on quantitative evaluation of corrosion and disbonding in metallic honeycomb structure. The principal value of thermographic inspection is that inspection times are short (e.g. 1m2 coverage in under 1 minute) DSTO has developed 'ThermView' processing software for qualitative image enhancement and for improved quantitative assessment, e.g. material loss due to corrosion damage estimated with better than 90% accuracy. [11]

8.4.7 Quantification of Peening Effects in Fatigue (G Clark and P K Sharp - AMRL)

Based on extensive research into the factors likely to influence fatigue in the RAAF's F/A-18 fleet, AMRL identified the peened surface condition of the 7050 aluminium alloy main structure as likely to be a dominant factor, particularly the competing effects of (a) beneficial residual stresses and (b) surface defects introduced by peening. Since there is considerable benefit in fatigue terms from minimising the surface damage, any reworks of the 7050 structure performed in Australia were done using tighter control of peening parameters than had been the case in original production. Significantly, in order to avoid simply driving existing surface damage deeper into the surface layers, any peening was to be performed only on a "clean" (ie. non-peened) surface.

This approach has been developed further into a means of local rework, in which a thin layer of material is removed (taking with it original damaged surface, and any associated fatigue cracking), followed by re-peening using well-controlled conditions. A critical part of the procedure is removal of the thin layer with appropriate confidence that all damage is being removed, and that no excessive cutting occurs. This rework method is currently being applied to full-scale fatigue test articles in Australia and Canada, as a means of reworking cracked areas, and allowing the test to proceed. It is also being used on fleet aircraft. [Results as presented in Figures 5 and 6]

An important part of the research supporting the life assessment of the RAAF F/A-18 fleet is estimation of the effect of original and reworked surface conditions on fatigue life ie the life improvement factor (LIF) for the various peening approaches. This is particularly complex, since samples of the fleet surface (and samples of the original material) are almost impossible to obtain, and it is necessary to rely on simulating the original condition as well as possible.

Several test series has revealed the following;

(a) The use of an Almen Intensity of 6A provided the longest fatigue lives
(b) Coverage of 200% (ie. saturation and then the same time again) provides best life.
(c) the run-out region of peening - a possible source of abnormal fatigue behaviour – has been demonstrated to be innocuous in preliminary tests. Initial results indicate that even a "sharp" edge to the peened area may not be a problem due to the large range of microstructural defects that initiate fatigue in 6-inch thick 7050 material. It is likely that the beneficial residual stresses in the peened layer extend into neighbouring unpeened material. Due to a large scatter in one test condition, the tests are being repeated..
(d) Current work is examining the potential use of the AMRL rework peening (ie. surface removal and re-peen) as a possible mid life rework method; initial results demonstrate that the approach may be used to re-gain full fatigue life on two occasions. Ultimately, of course, the likelihood of failure from other sources (e.g. internal defects, accidental damage, reduced cross-sectional area) will become more significant, and will limit the usefulness of the approach. However, the ability to extend life locally by a factor of two or three is almost always an adequate solution.
(e) For peening 7050-T7651, 6 Almen is the best intensity.
(f) Coverage should be keep at 200% as specified in PS14023.
(g) In nearly all cases peening gave an improvement in fatigue life over light hand polish.
(h) If a 6 Almen intensity and 200% coverage are used, a 1.4x life extension may be achieved with a 95% confidence. [12, 13]

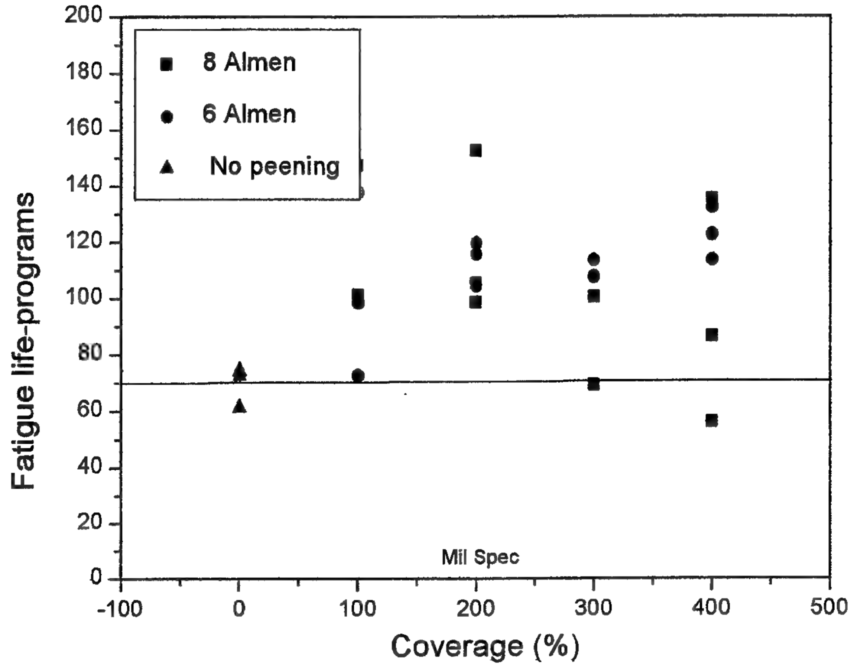


Figure 5: Effect of peening coverage and intensity on fatigue life.

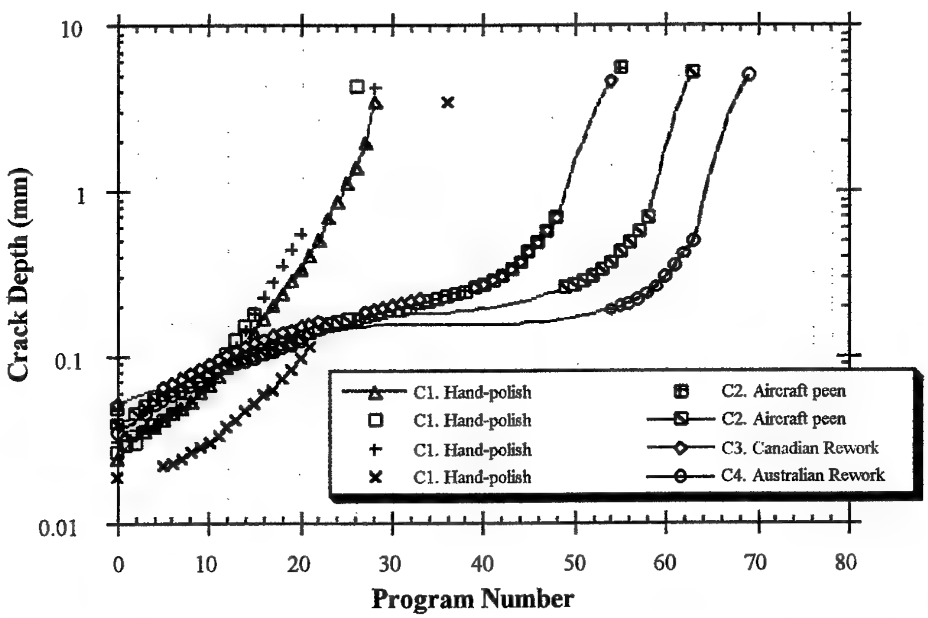


Figure 6: Crack growth from a peened surface. The two effects which control fatigue life for peened specimens are clearly visible: The retardation as the crack grows into the compressive residual stress field, , followed by a return to the unpeened growth rate once the crack tip is significantly beyond the minimum growth rate region.

8.4.8 Tearing fracture in Fatigue (G Clark, R Byrnes, P Khan Sharp)

Recent AMRL research assessed two empirical models for the stable tearing fracture which is observed on fatigue fracture surfaces. While it is correct to say that such tearing has little effect on overall fatigue life, it can be very significant if one considers that structural integrity often relies entirely on accurate estimation of the "crack growth life", and that large and unpredicted jumps in crack length can have a major effect on NDI intervals. No fatigue crack growth models currently address the occurrence of tearing, despite the fact that it may make up half of the fracture surface, even in relatively thick sections. Two tearing models, formulated by Schijve and Forsyth, have been assessed against test results, and both performed well as predictors of the stress intensity which causes tearing; both could be used to analyse existing fracture surfaces. However, over a range of specimen thicknesses, the usefulness of the models was limited, reflecting the difficulty of predicting tearing in thick section material, and the general difficulty of predicting the extent of the tearing which will occur in any given situation. [14, 15]

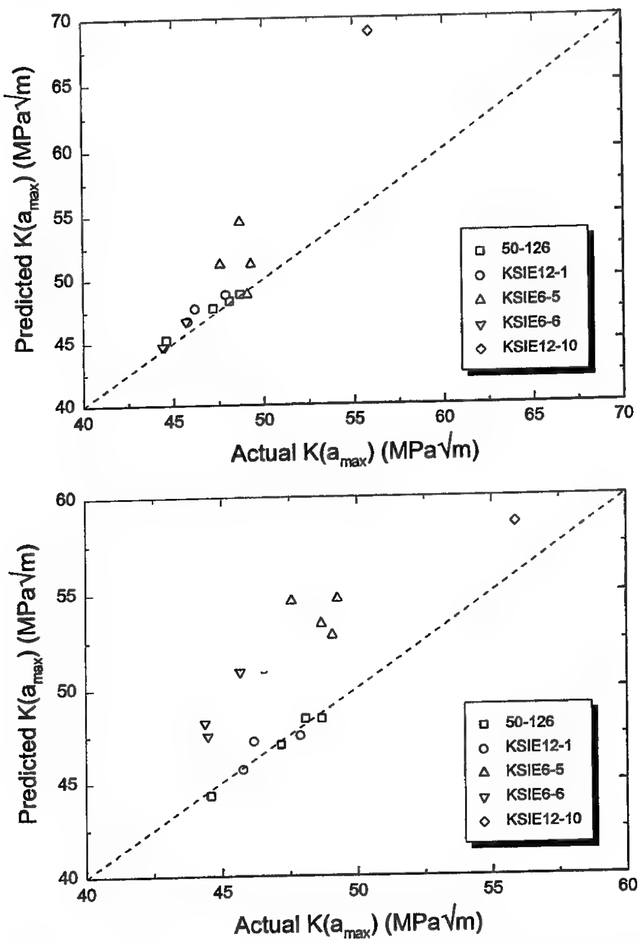


Figure 7: Trials of the tearing models of Forsyth (top) and Schijve (bottom), comparing the predicted and measured stress intensities after tearing for diferent thicknesses of specimen. Both models perform reasonably well for specimens in the plane-stress/plane strain transition range.

8.4.9 Reducing Scatter in Fatigue Testing (A F Cox and G Clark)

A recent series of fatigue tests on AF1410 steel coupons highlighted the benefit of assessing the effects of spectrum changes on fatigue behaviour by determining crack growth from the earliest detectable crack through the final failure. Where, until recently, 'life' has been used to compare fatigue behaviour under differing load conditions, albeit with the large inherent scatter in results, a better, more quantitative assessment of loading effects can be gained from crack growth studies in identified crack depth ranges. An example of this effect can be seen in the following graph, which illustrates the large variation in 'life' of the coupons under the two differing load regimes, but little variation in the crack growth rate from a depth of 0.25mm to failure depth.

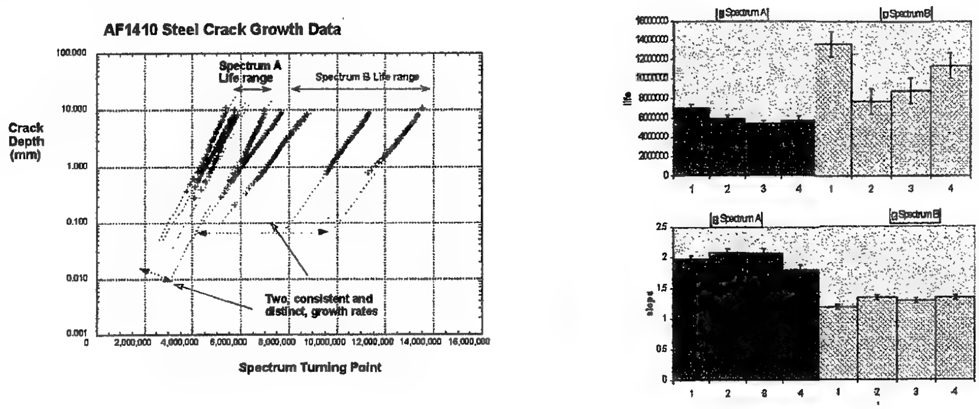


Figure 8: showing the reduction in scatter in the growth rate compared to the fatigue failure life, allowing improved discrimination between two fatigue conditions (in this case, spectrum type).

8.4.10 Fatigue crack growth; modelling vs reality – a case study (G Clark)

Recently, it was necessary to perform an analysis of the structural integrity of the wing of an aircraft which was lost in service; the analysis was complicated by the fact that the aircraft type is no longer in service, and no wreckage from the aircraft was available. Records from the fleet were naturally incomplete, and corporate memory of the various structural integrity management approaches then being introduced had largely been lost. Figure 9 shows recorded data.

The approach taken was to focus on the one aircraft, and its known condition. The aircraft was known to have cracks in the left wing spar, and concern had been raised that these cracks might have grown at such a rate that the then-current inspection intervals might have been inadequate. The analysis was performed to determine whether this was the case.

Given the now-incomplete fleet records and long-lost currency of our knowledge of test conditions etc., the obvious approach, namely re-estimation of a safe life or crack growth life for the fleet, was one which would inevitably lead to a highly conservative estimate of fleet life or interval. Such an estimate would not be useful as a yardstick against which to measure those developed and used by the RAAF.

Accordingly, the approach taken was to attempt to model the crack growth for the conditions in that wing as realistically as possible ie using best estimates of crack condition, geometry, growth etc. This estimate could then be compared to the then-current inspection interval to arrive at a conclusion as to whether or not the interval was appropriate for this particular case.

Several different models were used to estimate the crack size corresponding to the NDI records, and several different crack growth models – some simple, based on comparison with available data on crack growth from reworks, fatigue tests or crashed aircraft, and some complex, based on cycle-by-cycle crack growth analysis – were used. In all cases, the most realistic estimates of parameters were used where there was supporting information, although there were conditions where conservative assumptions had to be made through lack of any detailed knowledge concerning this particular wing.

The results indicated that the best estimate of crack growth for the wing was substantially slower than that required to cause failure within the approved flight conditions.

Interestingly, a means was later developed of analysing the incomplete fleet NDI records to extract data for similar-sized cracks throughout the fleet. This provided an estimate of fleet-wide crack growth for cracks similar to those in the subject wing, and provided an <u>independent and later</u> test of the best-estimate modelling . The results of the comparison are shown in Figure 10, and show that when allowance is made for different usage between this particular aircraft and the fleet, the modelled crack growth is slightly on the conservative side of the real fleet data, providing strong support for the original conclusions based on the modelling. [16, 17]

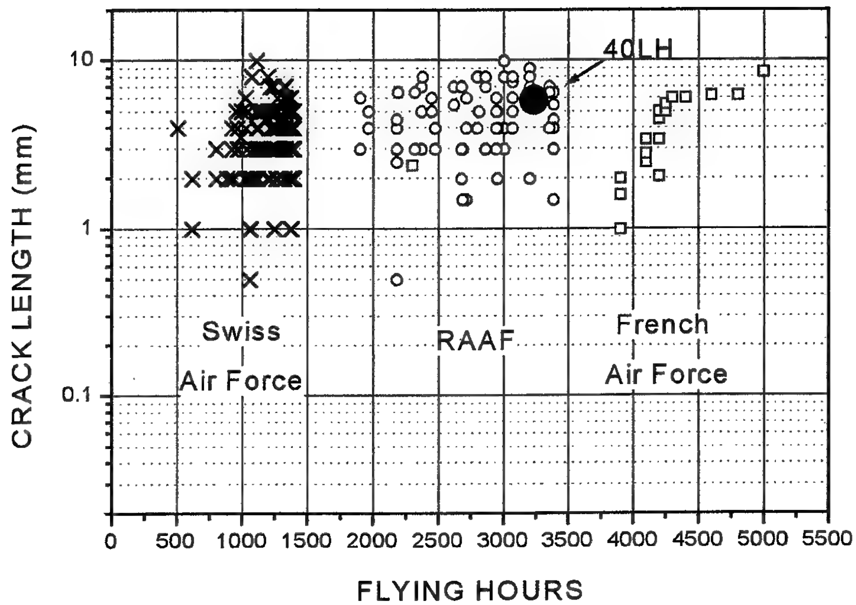


Figure 9: Snapshot of fleet crack surface length data for three different fleets, showing the subject wing in context. Note that data are not complete.

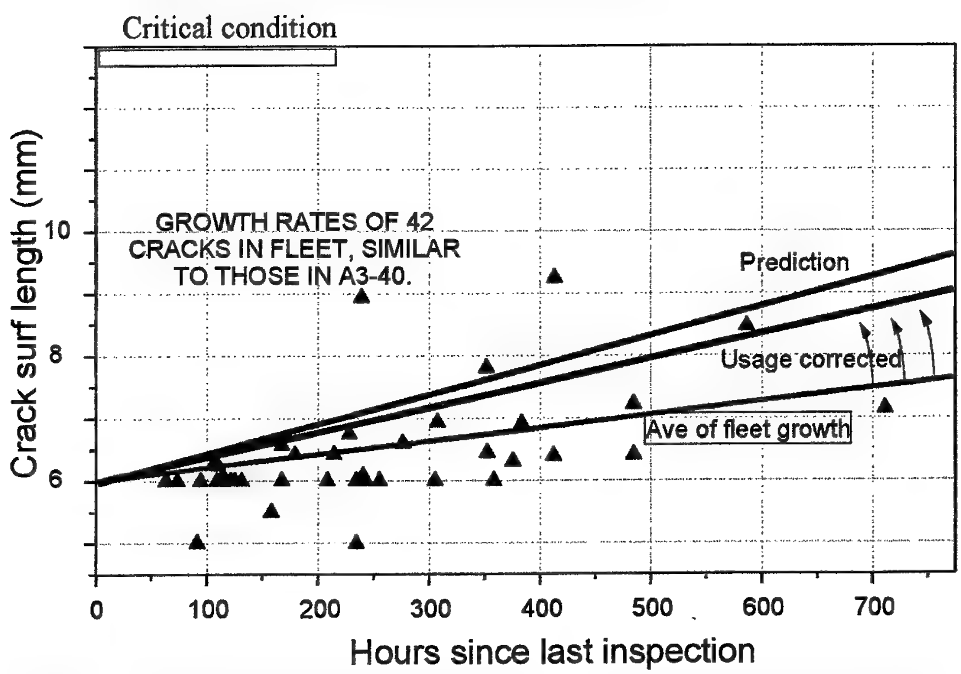


Figure 10: Comparison of the predicted growth of the cracks in the subject wing (based on best-estimate modelling), with later, independent data for fleet crack growth, corrected to the same usage rate as the subject aircraft. The results are very close, and reflect the remaining conservatism in the modelling process. The critical region for failure is indicated at top left.

8.4.11 Characterisation and Modelling of Multi Site Cracking Under Notch Plasticity Conditions (K. Walker AMRL)

A collaborative work program between AMRL and Lockheed Martin Tactical Aircraft Systems (LMTAS) in the USA is progressing in the area of Multi Site Cracking under notch plasticity conditions. The ultimate aim is to be able to model such behaviour which occurs in critical areas in the F-111 aircraft structure.

The collaboration with LMTAS includes developing and validating the computer program "METLIFE" which is designed to deal with fatigue crack growth in plastic notch fields. Coupon testing programs are also being carried out to produce data to correlate and validate the model. LMTAS have conducted tests on single edge notch specimens from D6ac steel material, and AMRL are testing similar specimens from 2024-T851 material.

The METLIFE program is based on a combination of local notch strain analysis and Linear Elastic Fracture Mechanics (LEFM) techniques. The program estimates the stress and strain distribution in the crack plane on a cycle by cycle basis, accounting for plasticity. The stress intensity factor is then calculated using a Greens Function approach and crack growth is estimated using standard LEFM techniques. The method has been demonstrated to predict cracking in situations where compression dominated loading induces significant notch plasticity and therefore residual tension stresses. Cracking is known to occur in service in the F-111 aircraft through this mechanism and the coupon tests have replicated the behaviour. Conventional fatigue crack growth modelling programs typically have difficulty in dealing with this situation. METLIFE is being enhanced to include a capability for multi site cracking and coalescence.

The 2024-T851 testing program at AMRL has produced valuable data for validating METLIFE. Constant amplitude and spectrum loading tests under tension dominated and compression dominated conditions have been carried out. The specimens have experienced significant amounts of notch plasticity (up to a maximum nominal stress of about 85% of the yield stress with a semi-circular notch present) and multi site cracking has been observed (see Figure 11). The crack formation and development has been accurately measured through the use of an advanced digital crack imaging system (Reference 18). These results will now be compared with METLIFE based analyses. The D6ac steel testing [19] yielded a limited amount of data and further testing has been recommended.

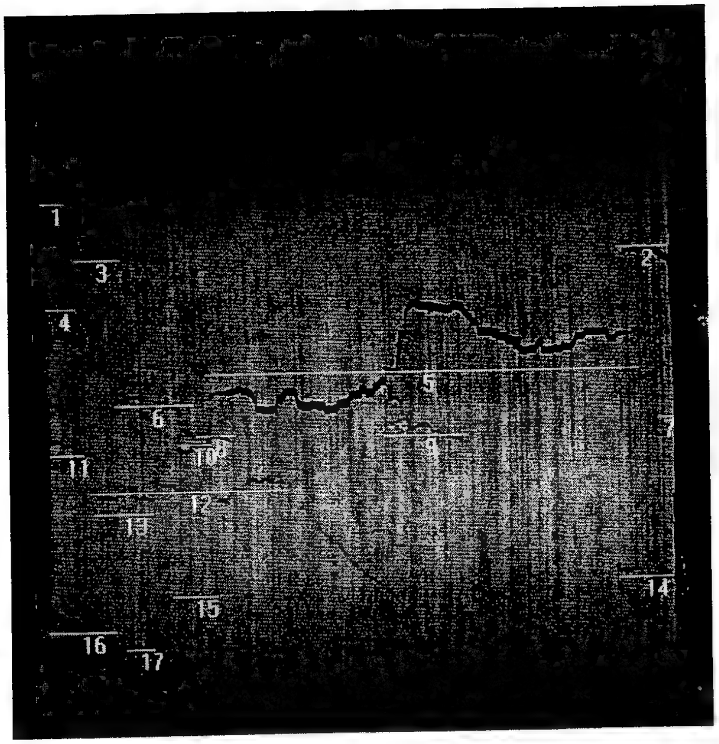


Figure 11: Multi Site Cracking on 2024-T851 Aluminium Specimen

8.4.12 Modelling of Short Fatigue Crack Growth under Cyclic Plastic Loading (C. H. Wang and L. R. F. Rose AMRL)

Cyclic plastic strain has long been recognised as one important factor resulting in the anomaly of short cracks growing faster than long cracks under the same applied stress intensity factor. Among the various analytical and empirical methods proposed so far, the most promising method has been the fully plastic *J*-integral theory by Hutchinson and his co-workers. In this investigation, a detailed elastic-plastic finite element analysis has been performed to examine the influences of non-proportional straining and large deformation on the near-tip deformation of a short crack embedded in a grossly plastic field [20]. It was found that existing theories would considerably under-estimate the crack-tip plastic blunting, as seen in Figure 12. This implies that those elastic-plastic correlating parameters for short cracks may be non-conservative in the prediction of short crack growth rates.

Based on the finite element results, an improved solution has been constructed for the crack-tip plastic blunting parameter. To account for the reduced plasticity induced crack closure for short cracks under cyclic plastic loading, a crack closure model has also been developed, which was shown to be in good agreement with experimental results [21]. With this aid of this new correlating parameter and the newly development crack closure model, a significant improvement has been obtained in correlating the growth rates of short cracks, as shown in Figure 13.

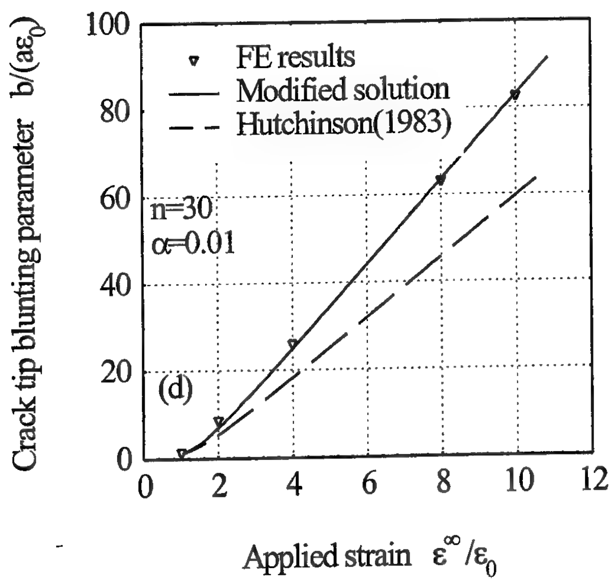


Figure 12: Variation of crack-tip plastic blunting size with increasing applied strain.

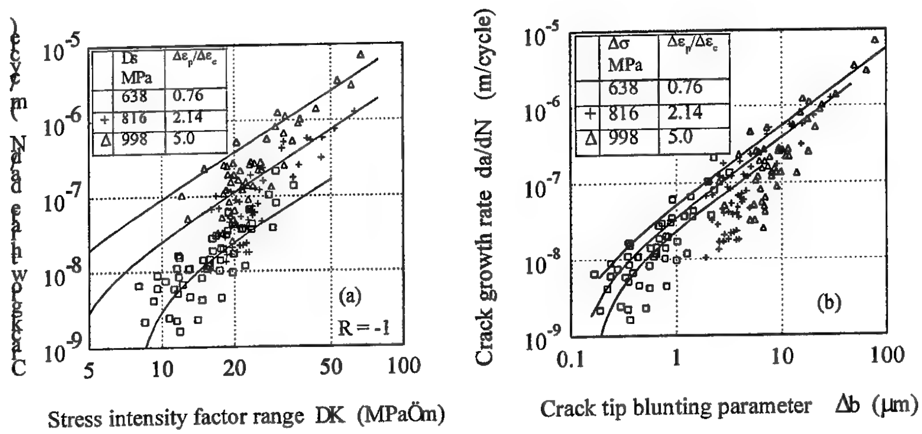


Figure 13: Crack growth rates for a medium carbon steel plotted in terms of (a) stress intensity factor and (b) crack-tip plastic blunting parameter.

8.4.13 Transient and Steady-State Deformation at Notch Root Under Cyclic Loading (C. H. Wang and L. R. F. Rose AMRL)

A problem has also been noted in a recent assessment of the methodologies used to predict the fatigue life of structural critical components in F/A-18 aircraft. This is primarily due to the inability of the Masing's assumption in calculating the relaxation of mean stresses at notch root under spectrum loading.

The transient and steady state cyclic deformation behaviour at a notch root is studied with a view toward assessing the rate of plastic shakedown of notched components subjected to cyclic loading. The failure of existing local strain approach, which is widely used for fatigue life assessment, to correctly account for the mean stress relaxation under asymmetric loading provides the main motivation of the work. Under localised plastic yielding condition, it is found that both elastic-perfectly plastic and linear kinematic hardening models predict an immediate plastic shakedown while nonlinear kinematic hardening law predicts progressive shakedown. At the shakedown state, linear kinematic hardening model predicts a non-zero mean stress whereas the nonlinear kinematic hardening model predicts a zero mean stress. To provide an efficient method for assessing the rate of mean stress relaxation under transient condition, an integral approach is presented for the rate of elastic or elastic-plastic shakedown at notch root [22]. As shown in Figure 14, the theory is in good agreement with finite element results.

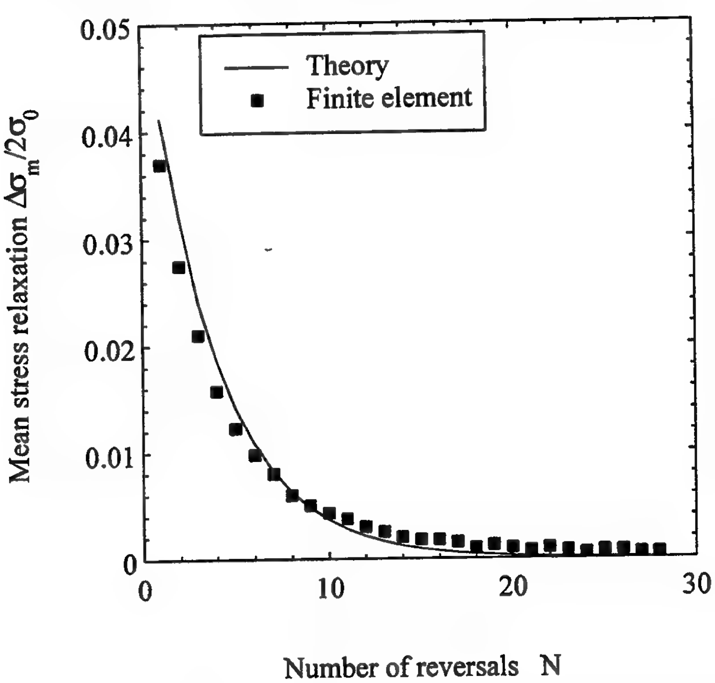


Figure 14: Relaxation of mean stress per reversal predicted using the integral approach

8.4.14 . Fatigue Crack Growth in Finite Thickness Plates (C. H. Wang and L. R. F. Rose AMRL)

For thin structures, fatigue crack growth rates may vary with the structure's thickness for a given stress intensity factor range. This effect is mainly due to the change in the nature of the plastic deformation when the plastic zone size becomes comparable with, or greater than, the cross sectional thickness. Variations in the constraint affect both the crack-tip plastic blunting behaviour as well as the fatigue crack closure level. Approximate expressions are constructed for the constraint factor based on asymptotic values and numerical results, which are shown to correlate well with finite element results. It is

demonstrated that the present results not only permit predictions of the specimen thickness effects on fatigue crack propagation under spectrum loading, but also eliminate the need to determine the constraint factor by curve fitting crack growth data; see Figure 15.

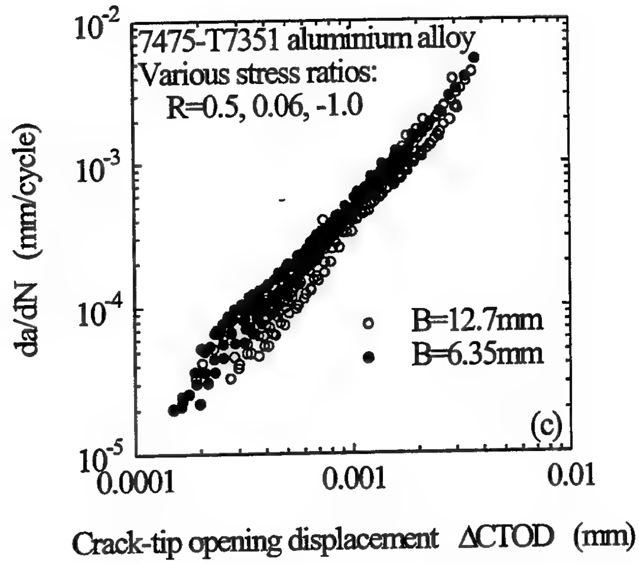


Figure 15: Crack growth rates obtained from specimens with two different cross-sectional thickness under various stress ratios.

8.4.15 Analysis of Fatigue Crack Growth from Cold-expanded/interference fitted Stop Drilled Holes (R.J.Callinan and C.H.Wang AMRL)

A theoretical investigation has been carried out into the fatigue life enhancement of a cracked plate specimen using a combination of stop drilling, cold expansion and interference fitting. Firstly the stress/strain state in the area in which cracks re-initiate from the cold worked hole is analysed using the ABAQUS finite element program. This involves an elastic-plastic and non-linear contact analysis. Validation of this work has been carried out using closed form solutions for plugs in holes for an infinite plate, corresponding to (1) cold expansion, (2) removal of the mandrel, (3) insertion of an interference plug, and (4) application of a remote load. It has been found that in the absence of an applied remote load the residual stress distribution in the area in which cracks would develop is approximately the same as would be expected in a cold expanded hole in an infinite plate. Also it has been found that under static tension loads use of cold working and interference plugs gives no additional strength, however results obtained from cycling of loads indicate that accumulation of strain energy per cycle is considerably less for cold worked and interference fit plugs, and hence a considerable improvement in life would be expected. Secondly, the introduction of short stationary cracks of crack-length of *0.066* to *1.0mm* has been considered with constraints to prevent crack closure. The crack-tip opening displacements of re-initiated cracks at the edge of the stop hole have been determined using the finite element method, and it is found the equivalent stress intensity factors are significantly higher, Figure 16, than those obtained by linear elastic fracture mechanics. From these results together with the FASTRAN II computer program, fatigue lifetimes are predicted which are in reasonable agreement with experimental data for these specimens.

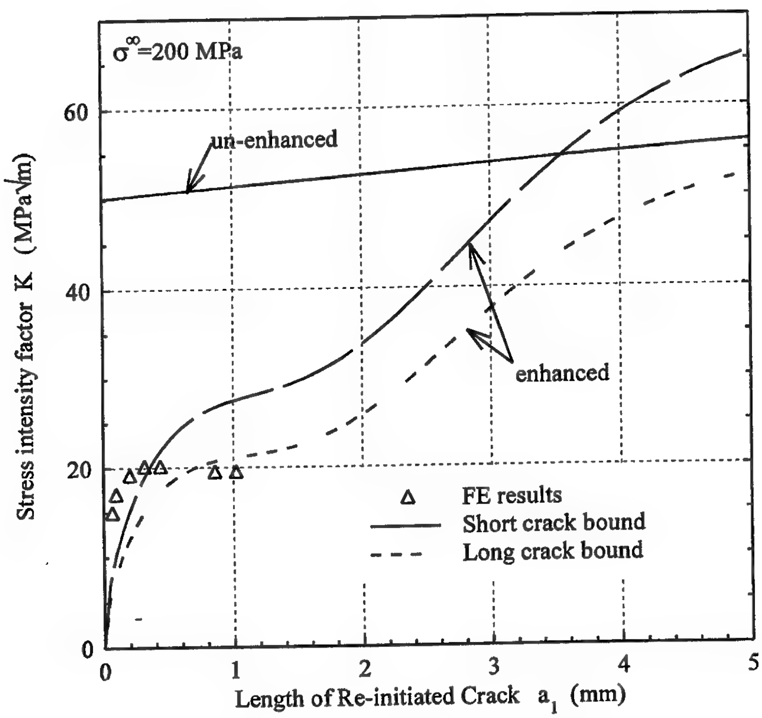
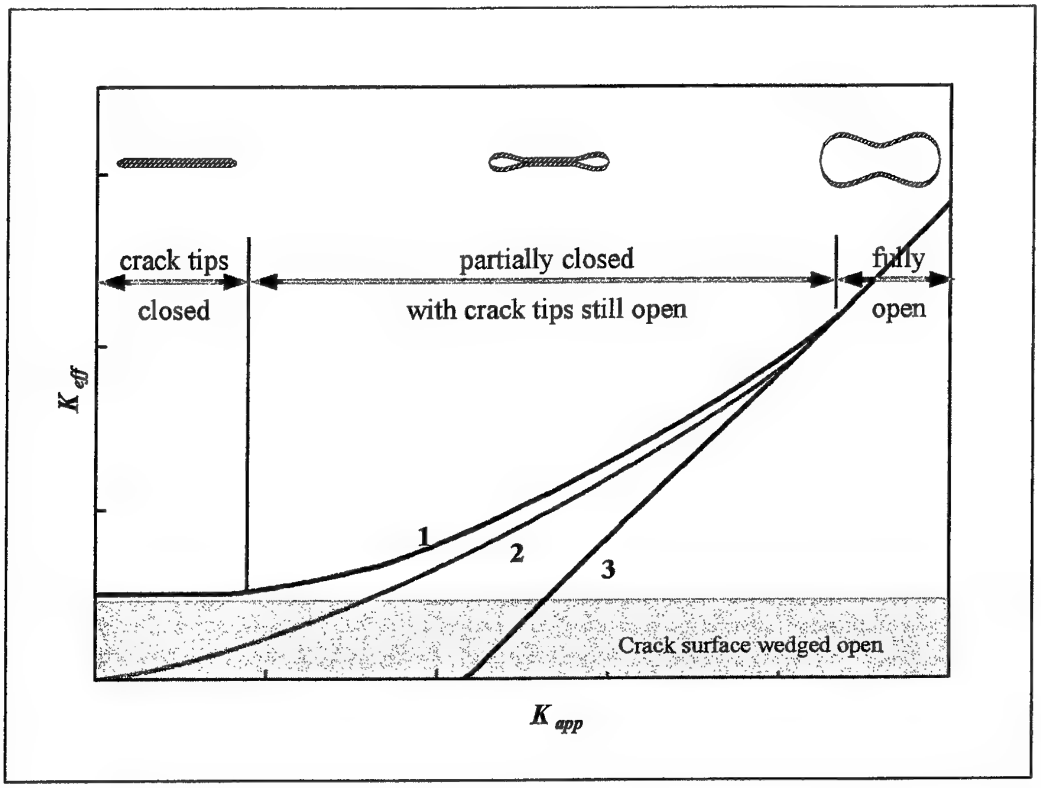


Figure 16: Stress intensity factor versus crack length for remote stress equal to 200 MPa.

8.4.16 Fatigue Crack Growth in Residual Stress Field: Implications of Partial Closure (X. Yu and A. Abel – The University of Sydney)

A modelling analysis has been performed to estimate the effects of residual stresses (RS) on fatigue crack growth [24]. In comparison with the traditional linear superposition approach, the model considers stress intensities contributed by partial crack closure forces. The interactions between the RS induced crack closure and the plasticity/oxidation induced crack closure have been considered. It has been shown that, in the existence of compressive RS field, the linear superposition approach may lead to inaccurate estimations at either conservative or non-conservative side. The necessity to consider the interactions between the RS induced crack closure and the plasticity/oxidation induced crack closures are also demonstrated.



Curve 1: estimations by the proposed model
Curve 2: estimations considering partial closure but ignoring plasticity/oxidation buildups
Curve 3: estimations based on linear superposition

Figure 17

8.4.17 Surface Layer Effect Model Of Fatigue Growth Of Small Cracks (Andrei Kotousov[1], John Price[1] and Simon Barter[2])

1. Monash University, Caulfield Campus, Mechanical Engineering Department
2. Defence Science and Technology Organization, Airframes and Engines Division

A theoretical model of the fatigue growth of small cracks is being developed. It is based on the surface layer effect that was first identified in the original Griffith's work on fracture. In our work it is assumed that this effect leads to an additional pressure that operates on curved surfaces. This pressure is very small at conventional sizes but it can be shown theoretically that it may play an important role at small radius of curvature like at a crack tip. The work has shown that an additional stress intensity factor due to the surface layer effect can be taken into consideration. This stress intensity factor is connected with mechanical properties of a material in a simple manner and has a clear mechanical meaning. It is dependent on the crack length as a root singularity and consequently rapidly decreases as the length increases and thus is insignificant at large crack lengths.

The model under development gives a reasonable qualitative explanation of the peculiarities of the kinetics of small crack growth. It is distinguished from numerous existing theories in that it does not introduce any correction factors, empirical equations, characteristic sizes or stages of small crack propagation and is size invariant. A comparison with experimental data on small fatigue crack growth has been carried out as shown Figure 18. [26]

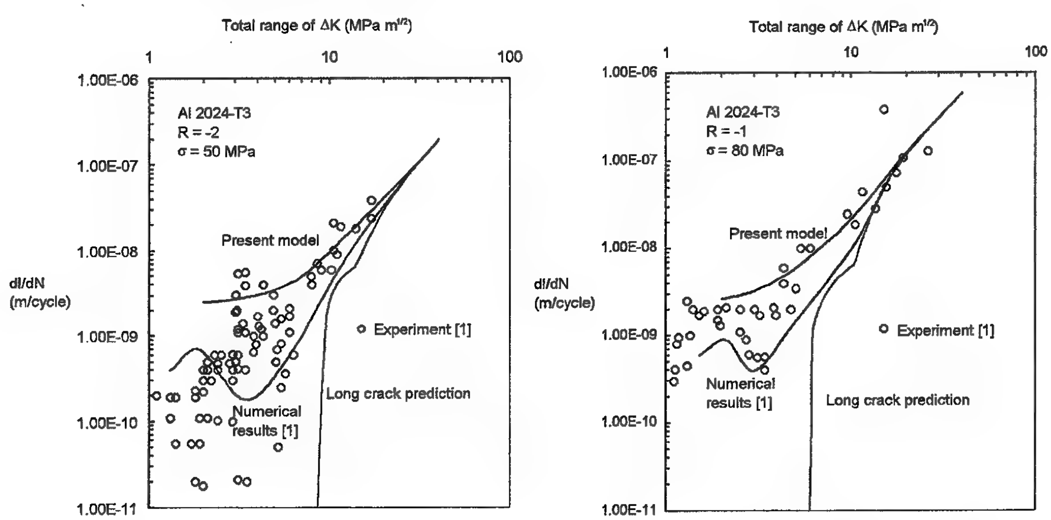


Figure 18: Comparison of predictions of the present model with numerical and experimental results for two different R values.

8.4.18 Reliability of Magnetic Rubber Nondestructive Inspection for Cracks in D6ac Steel (G. Hugo, C. Scala, C. Harding, H. Chin Quan – AMRL)

AMRL has recently commenced a program of work to determine the reliability of magnetic rubber inspections (MRI) performed on D6ac steel components on F-111. A relative unique feature of these inspections is the need to reliably detect, which a high degree of confidence, cracks smaller than 1mm in surface length and depth. The reliability of MRI will be characterized by an experimental program which will determine statistically the probability of detection (POD) of fatigue cracks in D6ac as a function of crack size. This information will be used as input to durability and damage tolerance analysis (DADTAs), which are used to determine the inspection intervals for in-service MRI of F-111 components. The experimental program will involve inspection, by Royal Australian Air Force technicians, of a set of laboratory specimens containing fatigue cracks of different sizes. The inspections will be conducted so as to simulate as closely as possible the conditions experienced by the technicians during in-service MRI of F-111 components. The experimental program will be completed during the period July 1999 to June 2000.

8.4.19 Influence of Out-of-plane Bending on One-sided Bonded Repairs (C. H. Wang and L. R. F. Rose AMRL)

Bonded repairs fall into two categories: two-sided (symmetric) and one-sided (asymmetric). In the former case two identical reinforcements are bonded on the two surfaces of a cracked plate. This symmetric arrangement ensures that there is no out-of-plane bending over the repaired region, provided the cracked plate is subjected to extensional loads only. In actual repairs, however, one-sided repair is often adopted in which composite patches are applied to only one side of the panel. This is because most often, only one face of a structure to be repaired is accessible and sometimes it may be permissible to patch only one side, e.g. aircraft fuselage or wing sections. Provided the structure to be repaired is well supported against out-of-plane deflection, the existing analytical procedure developed at DSTO has been shown to agree well with experimental and finite element results. However, in the case of an un-supported one-sided repair, the out-of-plane bending caused by the shift of the neutral plane away from that of the plate may considerably lower the repair efficiency, as recognised by a number of authors in the literature.

The aim of this investigation is to present a theoretical and numerical analysis of one-sided repairs, taking into account of the out-of-plane bending effect. To quantify the effect of out-of-plane bending, an extensive finite element analysis has been carried out to verify the analytical solution [27]. The most important finding is that the stress intensity factor of a one-sided repaired crack does asymptote to, but never exceeds, a limiting value, and that the analytical solution provides a *conservative* estimate for this limiting value.

The results of this work suggest that localised bending of the reinforcement in an un-supported one-sided repair would induce a significant increase in the stress intensity factor, hence reducing the repair efficiency. Nevertheless, the stress intensity factor has been found to approach, but never exceed, a limiting value with increasing crack length for the case of one-sided repair; see Figure 19. This means bonded repairs can be safely applied to un-supported structures, provided the efficiency of such a repair is quantified. In this work, an explicit analytical solution has been obtained for determining the repair efficiency [27, 28]. This estimate has been shown to agree well with finite element results and provide a conservative prediction suitable for design purposes.

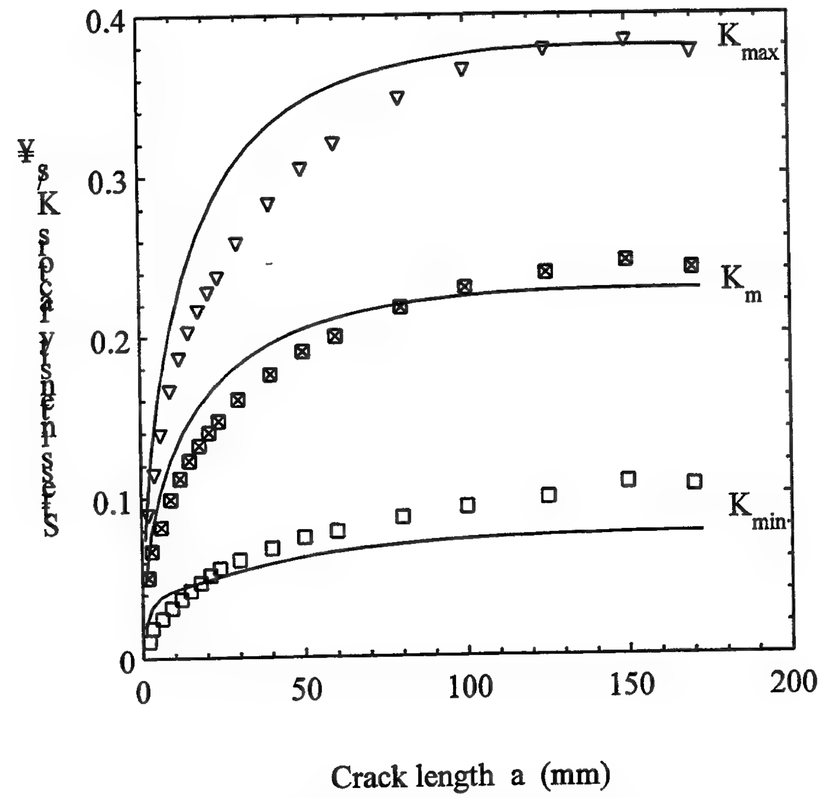


Figure 19: Theoretical prediction (solid curves) and finite element results (symbols) for a typical one-sided repair assuming geometrically linear deformation.

8.4.20 Efficiency of Bonded Patch Repair under Mixed Mode Loading (C. H. Wang and L. R. F. Rose AMRL)

The present composite patch repair technology has been mainly developed for mode I cracking. For instance, unidirectional boron composite patches are often aligned perpendicular to the crack so as to achieve the maximum repair efficiency. However, there are at least two circumstances where mixed mode cracking is a major concern in the context of bonded repair. Firstly, application of bonded reinforcements, which are frequently anisotropic, may alter the local stress-state near the crack region so that the maximum principal stress may no longer remain perpendicular to the crack plane. Secondly, structures are frequently subjected to non-proportional loading in which the principal stress/strain axes rotate with time, thus cracks may experience a time-dependent mixed mode loading. If the bonded repair technique is used to repair mode II cracks, one important question that remains to be resolved is whether this method is still effective. In particular, one finite element study [29] showed that the repair efficiency for mode II cracks was much lower than for mode I.

An analytical investigation was undertaken to determine the repair efficiency of bonded patch repairs under mixed mode load, especially mode II. By adopting the Fourier transform method by Keer *et al* [30], the problem of assessing the effectiveness of a bonded repair to a cracked plate subjected to mode II loading has been reduced to a one-dimensional integral equation, from which exact solution was then be obtained [30]. The results showed that repair efficiency for an isotropic under shear mode is lower than in the analogous mode I case (see Figure 20), and is strongly influenced by the shear stiffness of the patch.

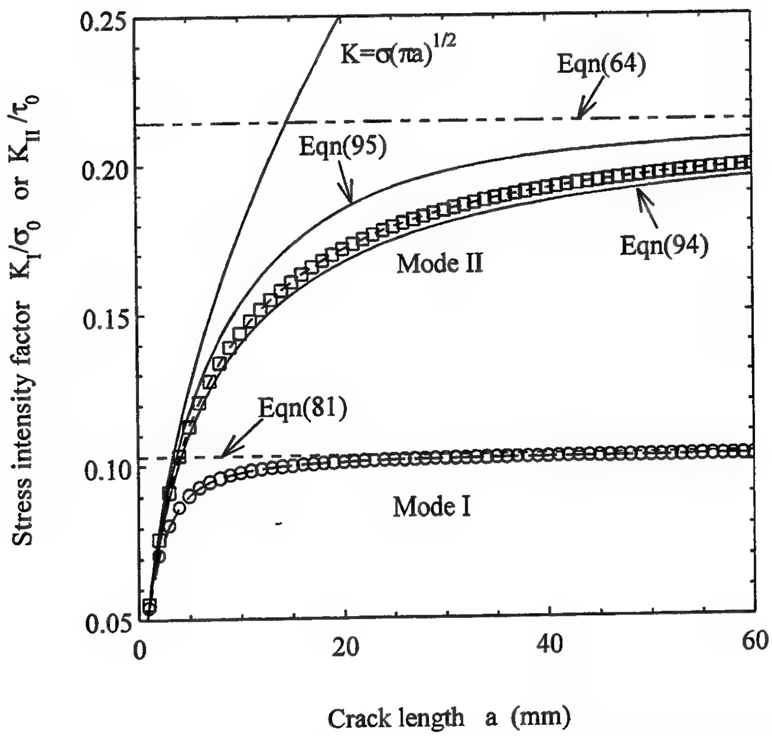


Figure 20: Normalised stress intensity factor for a isotropic bonded patch repair under mode I and mode II loading.

8.4.21 Life Extension Of An F/A-18 Aileron Hinge Using Structural Optimisation And Bonded Composite Reinforcement (R.J.Chester, M.Heller, S.Whitehead, R.Kaye DSTO And B.Teunisse Aerostructures)

F/A-18 inboard aileron hinges suffer from a fatigue cracking problem for which the only current solution is replacement of the component. A method has been developed for both reinforcing hinges that have not yet developed cracks as well as restoring cracked hinges to an airworthy condition. The approach developed is to use structural optimisation to reshape the hinge to reduce the stress concentration in the critical area. The rework profiles include an allowance for the removal of cracks if necessary. A boron/epoxy composite reinforcement can then be adhesively bonded over the reworked profile to further reduce the level of stress. This approach is able to reduce the stresses at the critical location by 16 % and a reinforced hinge has been tested satisfactorily to design ultimate load. Figure 1 shows the strain distribution along the aft strut of the hinge for three different hinge configurations. The unaltered hinge shows the peak in stress at the radius at around 67 mm. Following the 2 mm blend the stress peak broadens and shifts closer to the lug end. Reinforcement of the blended hinge with an eight ply boron composite reinforcement reduces the peak strain in the hinge as shown.

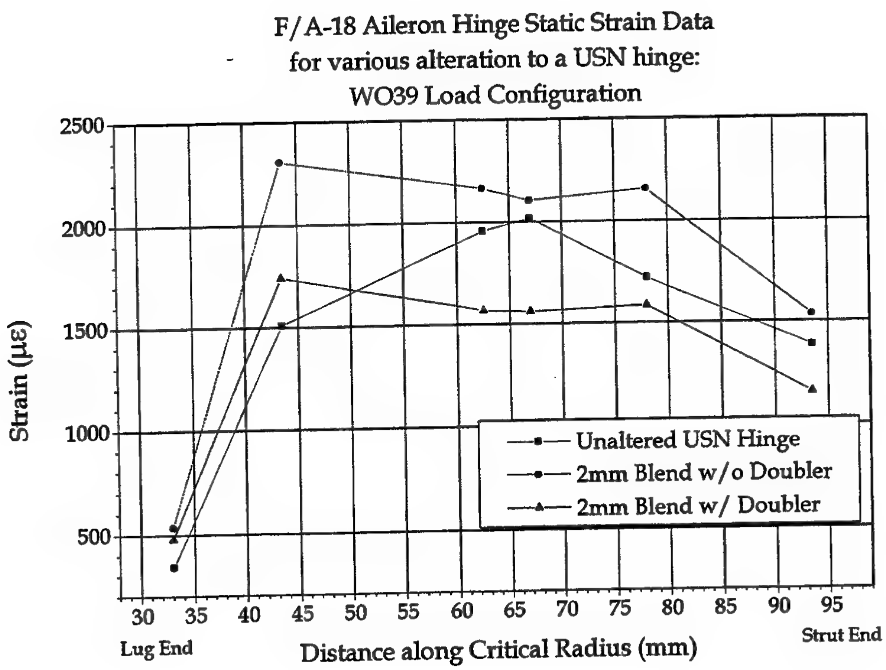


Figure 21: Strain behaviour of original, reworked and reinforced hinges.

8.4.22 Development Of Novel Nondestructive Evaluation (NDE) Techniques To Quantify Fatigue Cracking Under Boron-Epoxy Reinforcements To Complex-Shaped Aircraft Components (C. Scala and S.K. Burke – AMRL)

The application of composite reinforcement/repair technology to metallic aircraft components is being very actively pursued for the life extension of both military and civil aircraft components. The availability of suitable nondestructive evaluation (NDE) techniques to assess the effectiveness of such repairs is critical to the reliable application of the boron repair technology. NDE is required to monitor any fatigue crack propagation in the substrate following repair. While eddy current testing has been used routinely in the past to detect and monitor cracks under repairs, the fact that increasingly complex-shaped components are being considered as candidates for repairs is providing challenges in terms of NDE monitoring. This is particularly the case for repairs to curved surfaces such as recently proposed repairs to several F-18 components, including F/A-18 aileron hinges and a section of a CF-18 bulkhead.

Recent AMRL research has been aimed at developing NDE for crack monitoring under boron reinforcement/repairs to curved surfaces. Experimental results have been obtained on the application of a range of innovative eddy current and ultrasonic techniques to both the F-18 aileron hinge and bulkhead problems. The presence of curved surfaces made the application of conventional ultrasonics techniques difficult, due to the resulting complex wave propagation paths in the composite/substrate combination and the difficulty of maintaining probe coupling. However, a novel ultrasonic technique was developed which successfully detected very small cracks. The curved surfaces also complicated the application of eddy current testing, particularly in terms of controlling probe lift-off. The effect of variations in lift-off have been investigated in detail, particularly in relation to their effect on the sensitivity of eddy current techniques as applied to bonded repairs on curved surfaces. Possible solutions to this lift-off problem have been proposed.

8.4.23 Cold Expansion Tests for Plates Containing Elongated Holes. (M. Heller, R. Evans, and R. B. Allan AMRL)

Cold-expansion testing of D6ac plates containing an elongated (non-circular) hole has been undertaken using an AMRL designed interference fit plug/sleeve arrangement [32]. The aim has been to determine the practical viability of the process as an option for addressing the cracking problem at the non-circular fuel flow vent hole number 13 in the wing pivot fitting of the F-111C aircraft in service with the RAAF. Here three nominally identical plate specimens were cold expanded, with sleeves of two different material types (ie high strength tool steel (D2) and stainless steel) being trialed. The coupon geometry is shown in Figure 22 (however no remote stress was applied), while a schematic of the cold expansion plug/sleeve design is shown in Figure 23. Strain gauge readings around the elongated hole boundary were recorded at various stages of the cold-expansion process, as well as some typical full-field qualitative photoelastic strain distributions. In all cases nominal expansion levels greater than 2.5% were achieved, along with maximum peak strains of more than 10,000 micro-strain and maximum residual strains greater than 1600 micro-strain. These results indicate that highly effective cold expansion has been achieved. It was also demonstrated that subsequent to cold expansion, effective interference fitting could be achieved, without the need for post cold-expansion machining. The particular advantages available by using stainless steel sleeves have been identified, in that they were not damaged during the expansion process whereas the D2 sleeve was susceptible to cracking. Cracking of the plug was also evident and this was attributed primarily to the poor surface finish at the bore holes in the plug. Hence in future it is recommended that these holes have an improved surface finish. Fatigue testing of the existing design, (with minor amendments) has been recommended to confirm its anticipated suitability as an effective option for fatigue life extension. [32]

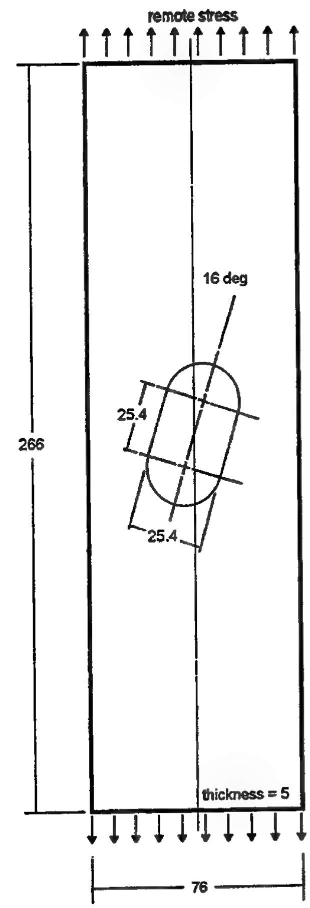


Figure 22: **Nominal geometry of test specimen plates for cold expansion testing**

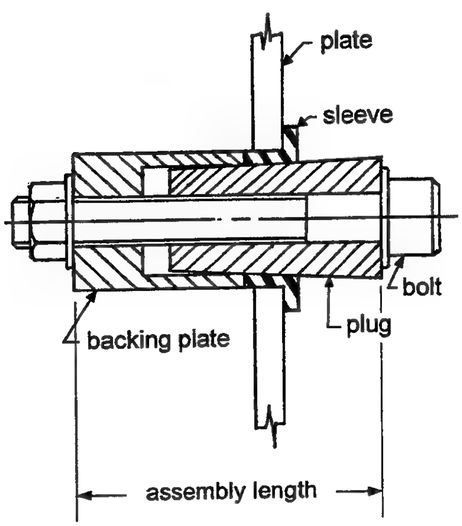


Figure 23: Schematic of the cold-expansion plug/sleeve design inserted into a plate with an elongated hole.

8.4.24 Fatigue Testing of Interference Fit Plug and Stress Bridge Life Extension Options for Coupons Containing Elongated Holes Representative of F-111C Fuel Flow Vent Hole Number 13
(M. Heller and R. Evans)

As indicated above, the development of a suitable interference-fit plug represents a possible option for the life extension of the non-circular FFVH#13, based on the results of computational and experimental stress analyses completed to date. Hence in the present work, fatigue testing has been undertaken for unenhanced and enhanced coupon specimens under representative spectrum loading conditions. Four coupon configurations were tested, namely (i) blueprint open hole, (ii) oversized open elongated hole, (iii) oversized elongated hole containing interference fit plug enhancement, and (iv) oversized elongated hole with stress bridge enhancement. Here the stress-bridge represents an alternative life extension option. All testing has been completed and is currently being documented [33]. Extensive data has been recorded, including non-linear strain responses (measured using strain gauges during (i) the cold expansion process, (ii) each CPLT loading, and subsequent in-flight loading. Also crack growth responses for at least one specimen representing each of the four main configurations has been completed using fractographic methods. In Table 1 approximate peak stress concentrations, determined from strain gauge measurements, are given for each of the four configurations. In Table 2 a summary of the fatigue test results is given, noting that one block is equivalent to 2000 flight hours. The overriding result is that the interference fit plug option effectively eliminated crack growth in the plate, and that it is an efficient and highly relevant option if required, for the life extension of the fleet and/or a planned F-111 wing fatigue test. Inspection of the components indicated that no damage occurred to the plug or the sleeve as a result of the expansion process or the subsequent fatigue testing. It was also found that the best stress-bridge configuration offers some life extension, however, the crack growth rate was approximately five hundred times greater than for the interference fit plug case, and its use induces multiple extensive cracking. [33]

Table 1: Relative static calibration strain results for FFVH13 coupon specimens

Enhancement case	Approx. normalised strain range (micro strain)	Approx. stress concentration
Blue print open hole	3700	5.1
Open elongated hole	3200	5.8
Stress bridge in elongated hole	2300	2.6
Inter. fit plug in elongated hole	950	0.9

Table 2: Fatigue test results for FFVH13 coupon specimens

Enhancement case	Fatigue life (blocks)	Failure type	Approximate Crack length (mm)	Residual strength (kN)
Blue print open hole				
FF13AB	23	Fatigue	16.4	
EM52AC2B	21	Fatigue		
Open elongated hole				
FF8AG	2	Fatigue	12.7	
EM50AC2B	2	Fatigue		
EM52AA2B	2	Fatigue		
Stress bridge in elongated hole				
FF9AG	31	Test terminated	30.9	196
FF10AB	31	Fatigue		
EM52AA1B	5	Fatigue		
Inter. fit plug in elongated hole				
FF8AJ	69	Test terminated		341
EM52AC1B	69	Test terminated		367
FF9AD	69	Test terminated	0.43	343
FF12AM	69	Test terminated		345

8.4.25 Smart Structures for Through Life Support Of Airframes (S. C. Galea - AMRL)

Smart Composite Bonded Patches

The "smart patch" approach, which is based on self-monitoring of the patch, is aimed at alleviating the certification requirements for implementing bonded composite repairs to primary aircraft structures [34, 35]. This approach relies on the ability to detect automatically disbonding in the patch, ie. the 'smart patch' approach is basically a continuous safety-by-inspection approach for the bonded repair. However, this approach brings its own problem of reliability assurance. Previous experimental work at the AMRL has shown that the concept of a patch health monitoring system by using ratios of (patch strain)/(strain in the component) appears to be quite promising. Therefore, the "smart" approach being studied in the AMRL is based on strain sensors bonded to the ends of the taper region and on the surface of the component away from the patch and the monitoring of the strain ratio during service life. Any decrease in this ratio is an indication of disbonding of the patch in this critical region. In this approach there is no requirement for measurement of the actual loading: disbonding is indicated by the reduction in relative strain.

The main aspects of the (remotely interrogated) patch health monitoring device were outlined in [34, 35]. Two devices were built and experimentally demonstrated on a laboratory specimen containing a boron/epoxy doubler using both resistance-foil and piezoelectric film sensors as the strain sensors. The results show that the patch health monitoring device was able to successfully detect and monitor damage in the safe-life region of the patch (i.e. the tapered region of the doubler). The latter sensors appeared to be extremely fatigue resistant and robust. However, this device has several areas needing further investigation and refinement. One area is the very difficult problem of interrogating the 'smart patch' through the aircraft fuselage/skin when the patch is situated in inaccessible internal locations. The second major area is to ensure all components (particularly the battery) perform properly over an extended operational temperature and altitude range. Although only one patch state-of-health value and two times are recorded on the prototype, further development of the electronics would allow a greater number of other parameters to be recorded as well.

Optical Fibre Sensors for Damage Detection in Bonded Repairs

Optical fibres offer a means of monitoring loads in a structure as well as the load transfer process in bonded repairs repairs, and can therefore be used to provide an indication of the integrity of the repair. This program of work, undertaken at Monash University, investigates the use of an array of Fibre Bragg Grating strain sensors (FBGs) for the in-situ health monitoring of bonded repairs and the associated structure.

Current work entails embedding studies of Bragg optical fibre sensors (OFS) in composite repairs/doublers of the TTCP double strap specimens (Figure 24). This work is part of a TTCP[1] operating assignment. The specimen has a 13 ply thick graphite epoxy doubler bonded to both sides of the specimen. Two different damage states have been induced in the specimen, viz: a disbond (teflon insert) in the adhesive layer and a disbond/delamination between the 2nd and 3rd ply of the doubler In this case the disbond extends for the full length of the taper. The aim here is to use OFS to detect damage in the safe-life zone of the doubler and then to monitor damage growth under fatigue loading. Also various sensors used by the other participants will be compared and evaluated. Experimental results will be compared with FEA results. FEA has been undertaken [36] on the specimen for a range of disbond/delamination lengths, viz: 0, 26, 31, 36, 41, and 46 mm, where the distance is measured from the

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[1] TTCP is an acronym for The Technical Cooperation Program, which is a collaborative program in Defence Science involving the UK, USA, Canada, Australia and New Zealand. This assignment was established by the Composites Technology & Performance Panel of TTCP.

edge of the patch. The effect that these had on the axial strain, when the specimen was subjected to a remote tensile stress of 137 MPa, was studied. To evaluate the relative merits of embedding versus surface mounted sensors the strains were computed on both the surface and in the 9th ply. The resultant strain profiles are presented in Figures 25 and 26, where x=0mm corresponds to the centre line of the specimen. From these figures it can be seen that, sensors located in the 9th ply, at about x = 30 mm, are more sensitive to small delaminations/disbonds. However, as the disbond length increases there is relatively little difference in the relative sensitivity of strains, as seen in the 9th ply or on the surface, to the delaminations/disbonds. This implies that, for the present problem where the sensors are located at x = 30 and 25 mm, there would be relatively little difference in sensitivity between surface and embedded strain sensors. On the other hand the figures also show that sensors located between 0 to 30 mm from the centre line will have great difficulty in assessing flaws less than 26mm in length. Furthermore, for the range of disbonds/delaminations under consideration, sensors located between 20 – 30mm from the centre line are, in general, more likely to be able to sense the change in the strain filed due to the disbond/delamination. For optimum detection of disbonds/delaminations in the taper, sensors should be located in the region of the taper.

Dynamics - Self-powered Vibration Suppression Device

There is considerable interest in the use of piezoelectric ceramic actuators for vibration suppression of thin lightly damped structures. Reference [37] looks at the concept of a self - powered discrete time piezoelectric damper that uses piezoelectric elements as the power source, sensor and actuator. The device uses a small portion of the electrical energy produced by the piezoelectric elements to power the electronic circuitry, thus eliminating the requirement for an external power source. The circuit controls the transfer of electrical energy between a storage device (capacitor) and the piezoelectric elements, which also operate as an actuator to suppress vibration. This device is referred to as a Strain Amplitude Minimisation Patch (STAMP) damper.

Initial experimental results have shown that a concept demonstrator of the STAMP damper had better damping than the simple resistor shunt damper but not as good as the tuned R-L resonant shunt damper (Figure 27). Reference [38] discusses the refinement of the STAMP technique in order to optimise the operation of the device. Analysis has shown the conversion efficiency of the PZT material (related to its dielectric loss and internal mechanical loss) and the power used in the control circuitry (predominantly in current switching components), are major factors in determining the amount of strain minimisation, available with this technique. By using a PZT actuator with a larger thickness of dielectric material to reduce dielectric losses and by using low loss switching arrangements to keep the electronic losses down, improved performance of the STAMP damper can be obtained. Work to-date has shown that for the PZT elements used here the STAMP device can theoretically, with ideal electronics (i.e. no losses) achieve an upper limit of 80% amplitude reduction. (Figure 28) However for PZT elements with lower losses (which are currently available), higher amplitude reduction is possible. The STAMP circuit does not need an external power supply or any large and heavy components, thus allowing it at a future date to be packaged with the PZT elements to form a simple to apply patch.

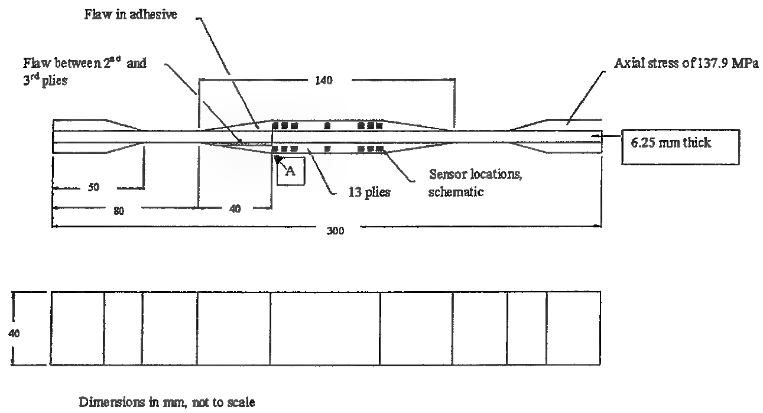

Figure 24: Geometry of the TTCP double strap test specimen.

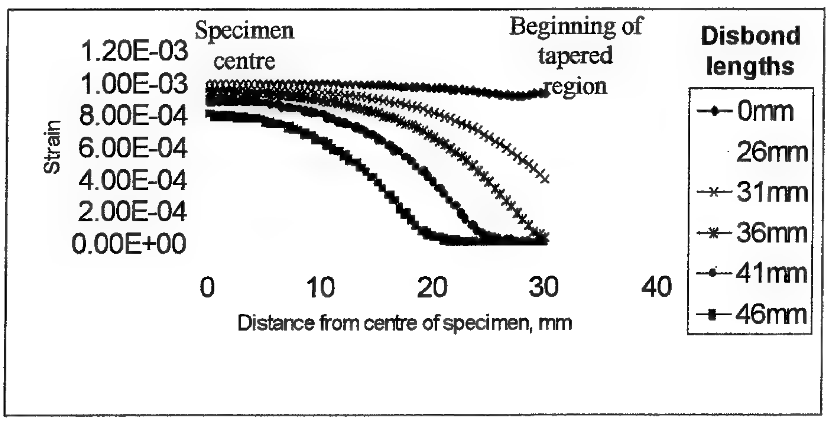

Figure 25: Strain in the 9th ply of the composite for various disbond lengths in the adhesive

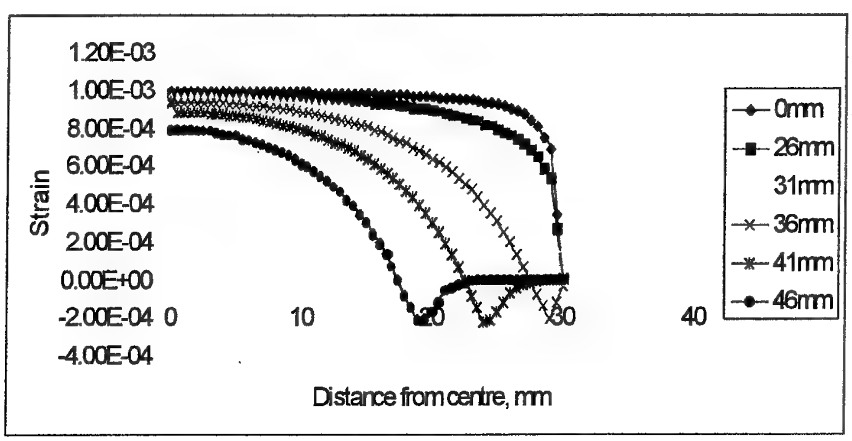

Figure 26. Strain on the surface of the composite for various disbond lengths in the adhesive.

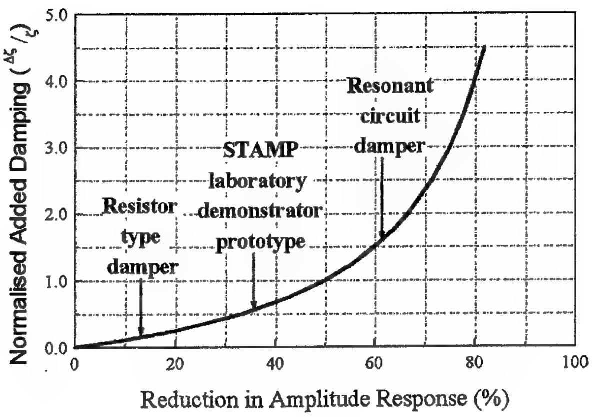

Figure 27: Added damping required to produce desired amplitude reduction.

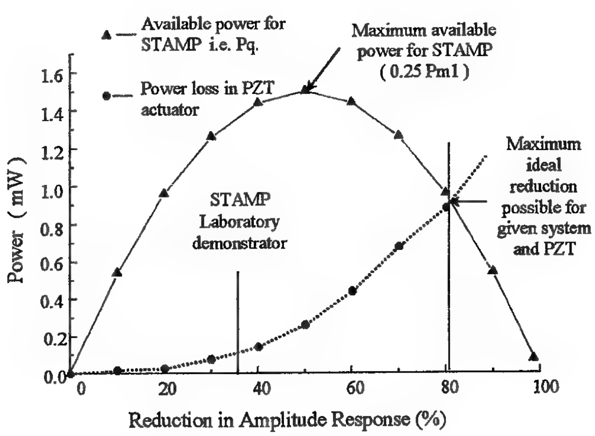

Figure 28: Power variation with amplitude reduction.

8.4.26 Determination Of The Stress Intensity Factor In Bonded Repairs To Acoustically Fatigued Panels
(R.J.Callinan, C.H.Wang, S.C.Galea, L.R.F.Rose and S.Sanderson)

The repair of cracked aircraft structures subject to in-plane loads using bonded repairs has resulted in considerable aircraft life time extension and hence cost savings [Baker and Jones]. However the use of bonded patches to repair panels with acoustically induced cracks (acoustic fatigue) is only recent and presents some difficulties. Acoustic fatigue is a result of high frequency lateral vibration of an aircraft panel as a result of time varying pressure waves caused by engine and/or aerodynamic effects. For example acoustically induced cracks have been recorded in the lower external surface of the nacelle skin of the F/A-18 aircraft, as shown in Figure 29. In this case the overall sound pressure levels of the order of 170 db have been recorded. Attempts to repair these cracks by applying standard methods of bonded repair were made, however the cracks continued to grow. These panels were repaired on the basis of in-plane loads. It is evident that the use of bonded repairs subject to acoustic fatigue requires analytical tools that take high frequency out-of-plane vibration into account. It has been found [Callinan et. al.] that while the boron fibre composite bonded repairs do reduce the stress intensity, the high number of cycles lead to significant crack growth. Since it is known that the amplitude of vibration is inversely proportional to the square root of the damping, then a combination of damping and stiffness may reduce the crack growth rate significantly. Highly damped repairs (Durability Patches) to acoustically damaged panels have been proposed by [Rodgers et.al.]. Furthermore [Liguore et.al] have applied a highly damped repair to the vertical fin of the F-15 aircraft, as part of a test program. In this work an analytical solution will be proposed to allow the computation of stress intensity in highly damped repairs, which involves the use of crack-bridging theory [Wang and Rose]. While crack bridging theory is applicable to statically loaded beams, the use of correction factors based on simple plate theory are applicable to rectangular panels.

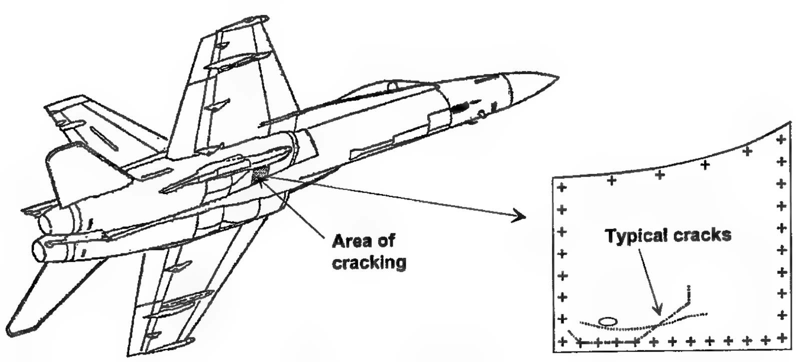

Figure 29: Location of the cracking in the lower nacelle inlet.

For practical cases of panels restrained on all edges it has been found the closed form solution always leads to conservative values of stress intensity. As a result this theory together with appropriate crack growth data, will enable the design of highly damped repairs of acoustically-induced cracked panels.

8.4.27 Buffet Load Alleviation (T Ryall AMRL)

Buffeting is an aero-elastic phenomenon, which plagues high performance aircraft, especially those with twin vertical tails like the F/A-18, at high angles of attack. High performance aircraft are, by their very nature, often required to undergo manoeuvres involving high angles of attack. Under these conditions unsteady vortices emanating from the wing and the fuselage will impinge on the twin fins (required for directional stability) causing excessive buffet loads, in some circumstances, to be applied to the aircraft. These loads result in oscillatory stresses, which may cause significant amounts of fatigue damage. Active control is a possible solution to this important problem. A joint program involving the USA, Canada and Australia was carried out in late 1997 and early 1998 to show that smart materials like piezoceramics under active control could be used to reduce stresses and hence extend fatigue lives. The work was carried out at AMRL's IFOSTP rig where simulated buffet loads could be applied to a full-scale structure. The experiments was shown to be extremely successful, the critical stresses were shown to be reduced anywhere between 50% and 4% depending upon the severity of the buffet. After taking usage rates into account this translated into an expected increase in life of 70%. Figure 30 shows the test article with the piezoceramics attached. [38, 39]

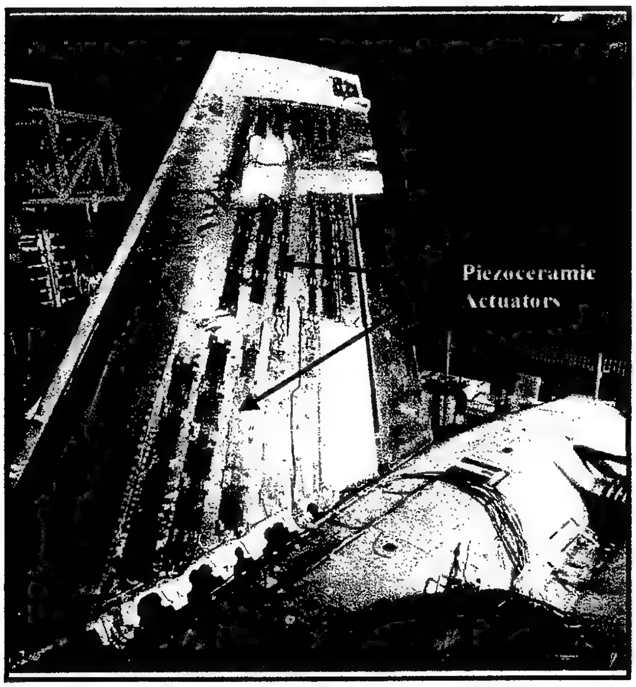

Figure 30: The test article with the piezoceramics attached

8.4.28 Risk and Reliability Analysis (David Graham AMRL)

Airframes and Engines Division, DSTO, have initiated a new task on structural risk and reliability analysis. The aim of this task is to establish a structural risk and reliability capability and re-establish a probabilistic fatigue analysis capability within AED, to provide support for RAAF fleet structural integrity investigations. The task will also review risk and reliability pertaining to engine and helicopter components and develop techniques for developing probability of detection functions associated with NDI techniques employed by the RAAF. A foundation for determining appropriate fatigue scatter factors utilising probabilistic theory will also be undertaken.

International collaboration in this field has been initiated in two areas. One is under a US Project Agreement on Aging Aircraft Research between DSTO and the USAF Wright Patterson Airforce Research Laboratory (AFRL), and the other is being initiated between TTCP countries. The latter consists of a round –robin assessment of existing probabilistic analysis packages.

8.5 REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN NEW ZEALAND 1997 – 1999

8.5.1 A4-K Skyhawk (P C Conor, S K Campbell – DOTSE)

An initial data review by W L Price [40] indicated the need for the RNZAF to determine the residual fatigue life remaining in its A4-K and TA4-K Skyhawk aircraft. The task of performing the Life of Type Study (LOTS) was let to contract.

SouthWest Research Institute (SwRI) was awarded the contract to perform the LOTS and associated Loads Environment Spectrum Survey (L/ESS). SwRI conducted an extensive data review for the RNZAF Skyhawk fleet. Based on the results of this review, SwRI recommended a number of critical structural locations that are most likely to determine the residual fatigue life remaining for the fleet.

SwRI had determined and documented the technical engineering methodology to be used in the LOTS and L/ESS, when the New Zealand government announced that they would be replacing the Skyhawk fleet with F-16 A/B aircraft much sooner than expected. The Skyhawk LOTS program was immediately halted as a direct consequence of the decision to purchase the F-16 fleet.

8.5.2 Networked Airframe Data Acquisition Recording System (NADARS) (G Rapley, S K Campbell – DOTSE)

Over recent years DOTSE has developed a fatigue data recording system known as NADARS. As reported in the previous ICAF review [41], the NADARS system consists of a number of small compact single channel digital recording modules. Each module records data from a single sensor (typically strain or acceleration) which is stored on non-volatile FLASH memory (0.5 MB per module). The individual recorder modules are connected to each other via a single twisted pair network, based on the CAN [42] protocol. The networked arrangement allows data to be extracted from all recorder modules from a single data milking point. It also allows a limited degree of signal synchronization to occur.

The prototype system was successfully trialed in one of the RNZAF Aermacchi MB 339CB aircraft. The trial has since been expanded to include the Skyhawk type. Clearance to use the NADARS system in the Skyhawk type is important for a number of future programs (as detailed in section 3.0 below).

8.5.3 A4-K Skyhawk Fatigue Monitoring (S K Campbell – DOTSE)

The initial work conducted on the Skyhawk LOTS (as reported in Section 1.0) highlighted three issues that require attention despite the full LOTS program being cancelled. In the mid 1980's the RNZAF instigated a major weapons and avionics systems upgrade program for their Skyhawk fleet. This upgrade added significant mass to the forward fuselage of the A4-K Skyhawk. Simultaneous with the avionics upgrade, the RNZAF refurbished the wings of their fleet. The wing refurbishment program involved replacing all three wing spars.

These two programs have significantly changed the structural configuration of the RNZAF Skyhawk fleet. The aircraft configuration used in all previous fatigue tests and calculations is no longer representative of the RNZAF fleet. The initial Skyhawk LOTS data review concluded that due to the configuration changes and other factors, there is sufficient uncertainty surrounding the fatigue life of three locations to merit detailed investigation. Although the full LOTS and L/ESS program has been canceled, DOTSE and the RNZAF have begun to clarify the structural issues surrounding these locations.

These investigations will involve using the NADARS system to record the strain history at two of these locations, as well as the Nz history for the aircraft. The two locations to be monitored are the cockpit

longeron (figure 31) and the nose attachment fittings (figure 32). These locations are of concern due to the significant increase in the mass caused by the avionics upgrade.

8.5.4 A4-K Skyhawk Intermediate Spar Fatigue calculation (S K Campbell – DOTSE)

Detailed in section 3.0 above, is a program to clarify a number of issues uncovered by SwRI in their initial work on the Skyhawk LOTS. The third critical location that is subject to uncertainty is on the Skyhawk intermediate spar (figure 33). Historically one of the life determining locations on the Skyhawk wing is a 2-inch hole in the intermediate spar near wing station 22.5 (WS 22.5). This hole is situated approximately 0.1 inch above the lower spar cap. The Skyhawk wing utilises an integral fuel tank, so this location is covered in sealant, and is typically "wet" with fuel.

DOTSE have determined that the stress field near this hole is extremely complex, and varies rapidly. The fatigue properties of this hole have been extensively investigated, however none of the analyses are in agreement. As this location is uninspectable, and any failure at this location would be catastrophic, DOTSE and the RNZAF have instigated a program to determine the consumed and residual fatigue life of the intermediate spar.

This program involves the determination of the strain field near the critical hole and the determination of the operational strain spectrum experienced by the hole. This data will be used to determine the life of the intermediate spar.

The strain field near the hole will be calculated by finite element modeling of the location. The location's geometry is extremely complex, with the hole being only 0.1 inch from the lower spar cap. There is a 0.1 inch fillet radius between the spar cap and the web, so the stress concentration due to the hole is multi-axial. The finite element models will be calibrated by static test of an uninstalled spar.

The NADARS system will record the strains in the Skyhawk wing skin directly below the critical location. It is assumed that there is a direct correlation between the strains measured in the skin and those developing in the spar cap directly beneath the skin. This transfer will be determined experimentally, and in conjunction with the finite element calculated stress fields, will be used to determine the strain spectrum experienced in service.

8.5.5 Soft Body Impact Damage on Composite Sandwich Panels (R R Aitken, D P W Horrigan-University of Auckland)

The preliminary results of this investigation were presented at the 1997 ICAF conference. The program involved studying the effects of soft body impact on composite sandwich panels. The program developed a model that predicted the extent of core crushing due to soft body impacts. The damage model included the effects of large plan area impacts on the residual load bearing capability of the panel. The effects on the life of the panel subjected to cyclic loading after soft body impact was also studied. There are plans to continue this work and further investigate the failure mechanisms of large plan area damage.

8.5.6 Analysis of Curved Anisotropic Sandwich Panels (D P W Horrigan – University of Auckland, in conjunction with Boeing Commercial Aircraft Corporation)

This study involved the determination of analytical stress calculations for a general curved anisotropic panel subject to various boundary and loading conditions. The work, performed in conjunction with the Stress Methods and Allowables Group of the Boeing Commercial Aircraft Company, produced a number of analytical solutions for certain loading and boundary conditions. The analytical solutions were found to be in close agreement to numerical solutions developed using finite element modeling. The work is being expanded to cover further boundary and loading conditions.

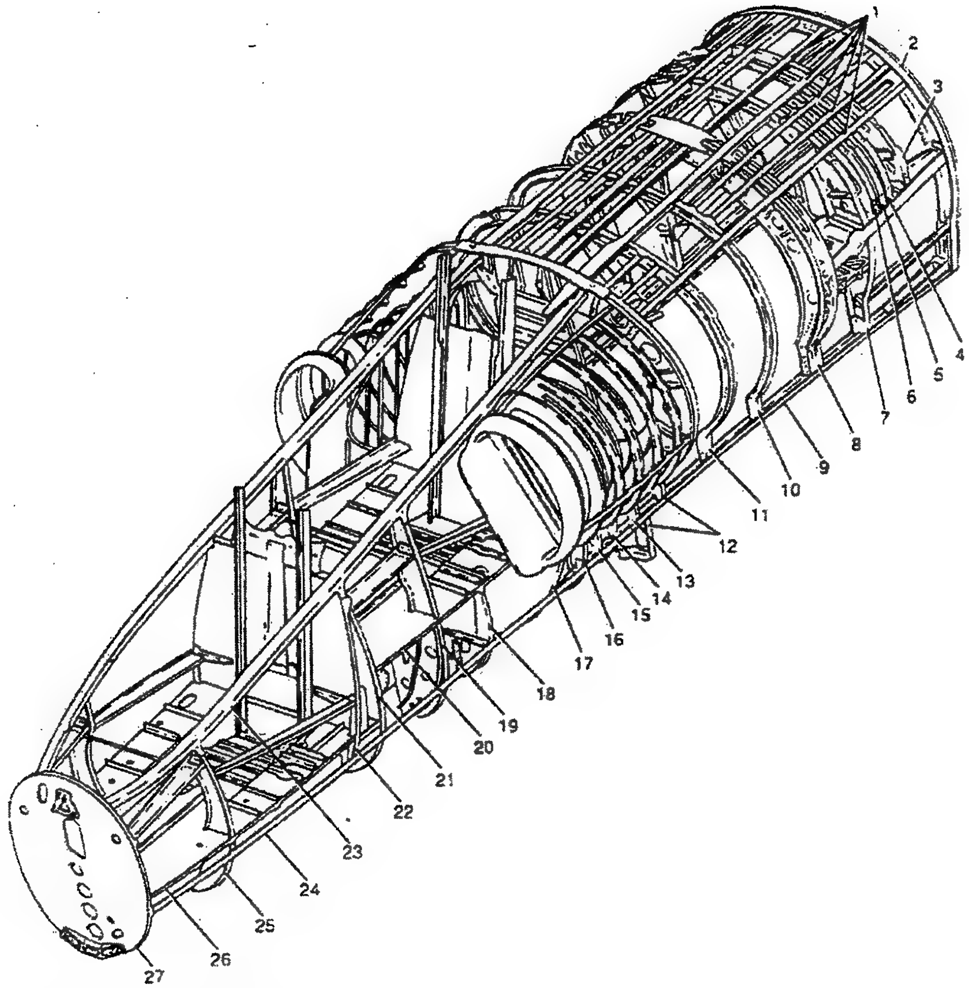

Figure 31: Skyhawk Forward Fuselage. The critical longeron is labeled 23.

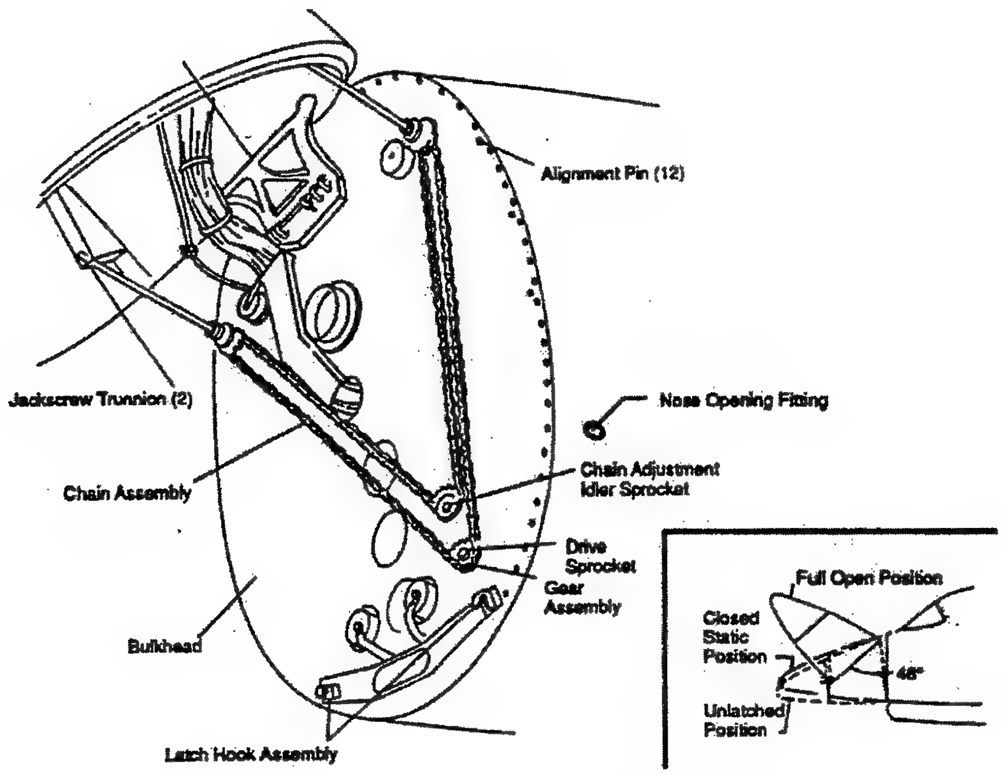

Figure 32: Skyhawk Nose Section showing the critical attachment fittings.

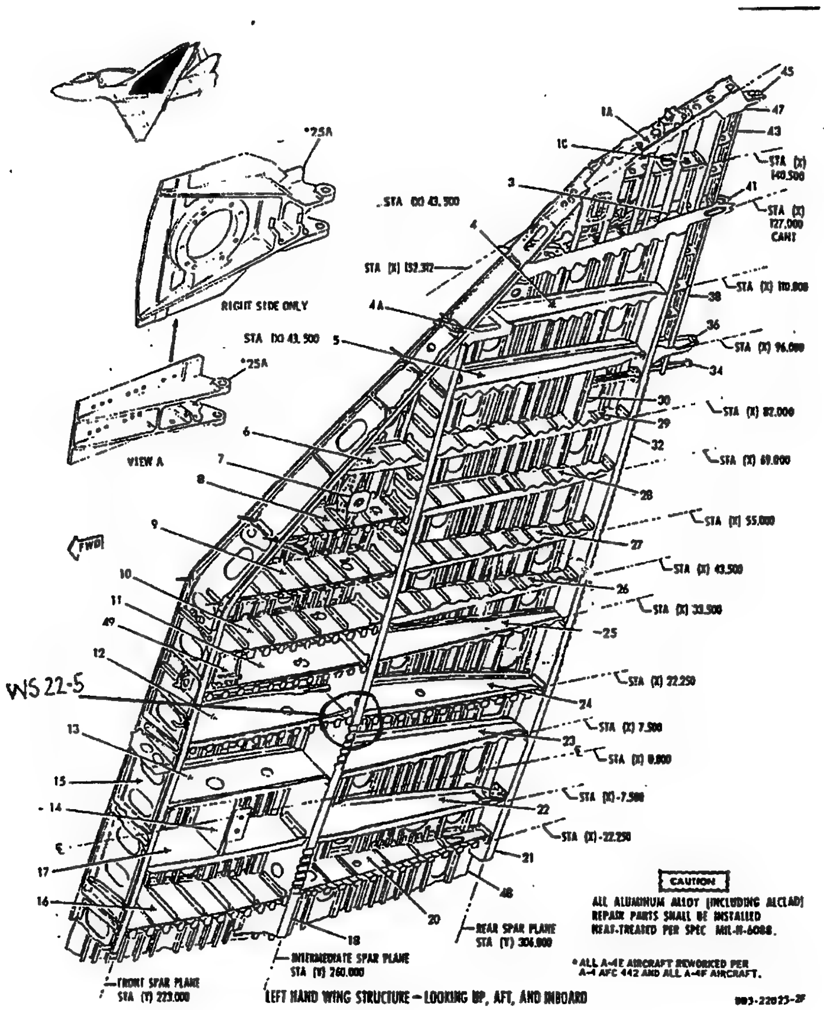

Figure 33: Critical Location on Skyhawk Intermediate Spar

REFERENCES

1. Molent, L., "Proposed Specifications for an Unified Strain and Flight Parameter Based Aircraft Fatigue Usage Monitoring System", USAF ASIP Conference, San Antonio, TX, Dec 1998.
2. Molent, L., "A Unified Approach to Fatigue Usage Monitoring of Fighter Aircraft Based on F/A-18 Experience", Proc. ICAS 98, Melbourne, Aust., Sept., 1998.
3. Molent, L., "A Review of a Strain and Flight Parameter Data Based Aircraft Fatigue Usage Monitoring System", Proc. of USAF Structural Integrity Prog. Conference, San Antonio, TX 3 - 5, Dec., 1996.
4. Aktepe, B and Molent, L., "Management Of Airframe Fatigue Through Individual Aircraft Loads Monitoring Programs", Proc. Inter. Aerospace Congress, Adelaide, Aust. Sept 1999.
5. Molent, L. and Ogden, R. (editors), "Review of RAAF F/A-18 "Structural Appraisal of Fatigue Effects (SAFE)" Version 112, Vol 1 - Detailed Review", DSTO-TR-0381, Jan., 1998.
6. Graham, D. and Molent, L., "Review of RAAF F/A-18 "Structural Appraisal of Fatigue Effects (SAFE)" Version 112, Vol 3 - Overview", DSTO-TR-0381, Dec., 1997.
7. Molent, L., Graham, A. D. and Aktepe, B., "Buffet Loads And Structural Redundancy Considerations In Vertical Tail Fatigue Monitoring Programs", Proc. Inter. Aerospace Congress, Adelaide, Aust. Sept 1999.
8. Jones,R, Knopp, M, Price, J. and Molent, L, "Stress And Strain Estimation At Notches In Aircraft Structures", DSTO-GD-0196, Meboume, Aust, Oct. 1998.
9. Agius, J., Connor, P., Madley, W. and Watters, K., *Royal Australian Air Force (RAAF) F-111 Aircraft Structural Integrity Sole Operator Program*, Proceedings of 1998 USAF ASIP Conference, San Antonio, Texas, USA, 1st to 3rd DEC 1998.
10. Sharp P.K, Cole G. and Clark G., *The Influence of Corrosion on Aircraft Structural Integrity*, International Council of the Aeronautical Sciences, ICAS 98, Melbourne, September 1998
11. N Rajic, Report in preparation Defence Science and Technology Organisation.
12. "The Effect of Peening on The Fatigue Life of Aluminium Alloys" P K Sharp and G Clark, Proc 9th International Conference Fracture, Sydney, 1-4 April 1997, pub Elsevier pp1491-1500.
13. "The Life Improvement factor for Peening of 7050 Aluminium Alloy." P K Sharp, G Clark, Report in preparation, Defence Science and Technology Organisation.
14. Byrnes R, Sharp P.K, Goldsmith N.T. and Clark G., *Empirical Analysis on Stable Tearing on fracture Surfaces*, Structural Integrity and Fracture Conference, Australian Fracture Group, 21-22 September 1998
15. Sharp P.K., Byrnes R, Goldsmith N.T. and Clark G., *Modeling of Stable Tearing During Fatigue of 7050 Aluminium Alloy*, DSTO-TR-(In preparation), October 1998
16. "The Effects of Cracks Identified in Mirage Wing 40LH on Wing Structural Integrity" G Clark, DSTO-DDP-0209, Defence Science and Technology Organisation, Dec 1997.
17. "The Effects of Cracks Identified in Mirage Wing 40LH on Wing Structural Integrity" G Clark, DSTO-TR-0771, Defence Science and Technology Organisation, Feb 1999.
18. Walker, K.F., Radtke, T.C., Kowal, E.A.B., and Bettoni, A., "Crack Imaging and Multi Site Crack Initiation", to be presented at the 8th International Aerospace Congress, Adelaide, Australia, September 1999.
19. "F-111 Wing Pivot Fitting Multiple Site Crack Initiation (MSCI) Study", FZS-12-640, LMTAS, USA, 31 December 1998.

20. Wang, C. H. and Zhao, M. (1999) Crack tip deformation under grossly plastic conditions, *International. Journal of Fracture* (accepted).

21. Wang, C. H. and Rose, L. R. F. (1998) Crack-tip plastic blunting under gross-section yielding and implications for short crack growth, *Fatigue and Fracture of Engineering Materials and Structures*.

22. Wang, C. H. and Rose, L. R. F. (1998) Transient and steady-state deformation at notch root under cyclic loading, *Mechanics of Materials*, Vol. **30**, 229-241.

23. Guo, W., Wang, C. H. and Rose, L. R. F. (1999) On the influence of cross sectional thickness on fatigue crack growth, *Fatigue and Fracture of Engineering Materials and Structures* (in press).

24. Yu, X. and Abel, A. (1999) Fatigue crack growth in residual stress field: implications of partial closure, presented in: The Second Australasian Congress on Applied Mechanics, ACAM99.

25. Short-Crack Growth Behavior in Various Aircraft Materials, AGARD Report No. 767, NATO Advisory Group for Aerospace Research and Development, 1988.

26. Kotousov A.G. and Price J.W.H., On Propagation of Small Surface Breaking Cracks, International Journal of Fracture, in print.

27. Wang, C. H., Rose, L. F. R. and Callinan, R. (1997) Analysis of out-of-plane bending in one-sided repair, *Int. J. Solids and Structures*, Vol.**35**, No.14, pp.1653-1675.

28. Wang, C. H. and Rose, L. R. F. (1999) A crack bridging model for bonded plates subjected to tension and bending, *Int. J. Solids & Structures*, Vol.**36**, 1985-2014.

29. Ching-Hwei Chue, Li-Chung Chang and Jang-Shing Tsai, (1994) Bonded repair of a plate with inclined central crack under biaxial loading, *Composite Structures*, Vol.**28**, pp.39-45.

30. Keer, L. M., Lin, C. T., and Mura, T. (1976) Fracture analysis of adhesively bonded sheets, *Journal of Applied Mechanics*, Vol.**98**, No.4, pp.652-656.

31. Wang, C. H. and Rose, L. R. F. (1998) Bonded repair of cracks under mixed mode loading, *International Journal of Solids and Structures*, Vol.35, 2749-2773.

32. Heller, M., Evans, R. E., and Allan, R.A., *Cold expansion tests for plates containing elongated holes*, AMRL Technical Note, file M1/9/318, 1999.

33. Heller, M., and R. E. Evans, *Fatigue testing of interference fit plug and stress bridge life extension options for coupons containing elongated holes representative of F-111C fuel flow vent hole number 13*, DSTO Technical Report, file M1/9/610, 1999.

34. Baker, A.A., Galea, S.C., and Powlesland. I.G. 1998 Presented at the Second Joint NASA/FAA/DoD Conference on Aging Aircraft, Williamsburg, Virginia. August 31 – September 3, 1998. A Smart Patch Approach For Bonded Composite Repair/Reinforcement Of Primary Airframe Structures

35. Baker, A.A. 1999 Presented at International Committee on Aeronautical Fatigue (ICAF 99) Washington USA 12-16 July, 1999. Issues in the Certification of Bonded Composite Patch Repairs for Cracked Metallic Aircraft Structures

36. McKenzie, I, Chiu, W.K., Jones, R. and Galea, S.C. COE-SM-97-03, Centre of Expertise For Structural Mechanics, Department of Mechanical Engineering, Monash University, August 1998. (Final Report) Health Monitoring of Bonded Repair Systems.

37. Konak, M.J., Powlesland, I.G., Van der Velden, S.P. and Galea, S.C. Proceedings of the 2nd Australasian Congress on Applied Mechanics, Canberra 10-12 February, 1999. Optimization Of A Self-Powered Discrete Time Piezoelectric Vibration Damper

38. Hopkins, M., Henderson, D., Moses, R., Ryall, T., Zimcik, D., and Spangler, R., "Active Vibration Suppression Systems Applied to Twin Tail Buffeting," SPIE 5th Annual International Symposium on Smart Structures and Materials, SPIE Paper No. 3326-05, San Diego, CA, 1-5 March 1998.

39. Ryall, T., Moses, R., Hopkins, M Henderson, D., Zimcik, D & Nitzsche F., "Buffet Load Alleviation" 2nd Australasian Conference on Applied Mechanics Canberra Australia 10-12 February 1999

40. Price, W L, *RNZAF A-4K Loads Environment Spectrum Survey L/ESS – Preliminary Data Review,* DSE Technical Note 95/3, May 1995

41. Poole P *Ed, ICAF 97 National Reviews,* DERA, Farnborough

42. ANSI/ISO Standard 11898-1993, *Road Vehicles – Interchange of Digital Information – Controller Area Network (CAN) for High-Speed Communication.*

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4. AUTHOR(S)	5. CORPORATE AUTHOR
Colin Martin	Aeronautical and Maritime Research Laboratory PO Box 4331 Melbourne Vic 3001 Australia

6a. DSTO NUMBER	6b. AR NUMBER	6c. TYPE OF REPORT	7. DOCUMENT DATE
DSTO-TN-0206	AR-010-988	Technical Note	June 1999

8. FILE NUMBER	9. TASK NUMBER	10. TASK SPONSOR	11. NO. OF PAGES	12. NO. OF REFERENCES
M1/8/1233	AE/MGT	DSTO	58	42

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19. ABSTRACT

This document has been prepared for presentation to the 26th Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Bellevue, Washington USA on 12th and 13th July 1999. The review covers fatigue–related research programs as well as fatigue investigations on specific military and civil aircraft in research laboratories, universities, and aerospace companies in Australia and New Zealand during the period April 1997 to March 1999.